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This report involves the investigation, analysis, and cost optimization of conventional two-stage tandem booster systems which utilized state-of-the-art advanced concept propulsion systems in the 0.6 to 3 million pound thrust range. Numerous configurations utilizing various combinations of engine types, thrust ratings, and propellants were considered and optimized for both performance and costs. The most promising configurations in the 600 K and 1500 K thrust classes for both state-of-the-art and advanced propulsion systems were adopted for investigation of 800 K and 3000 K thrust booster systems.

Various alternate booster system concepts were also investigated and evaluated on a cost comparison basis with the selected two-stage booster system configurations incorporating advanced high chamber pressure engines.

The basis for evaluation and comparison of the various configurations and concepts was dollar cost per pound of payload in a 300-nautical-mile orbit. In order to determine economic trends influenced by varying launch rates, costing was based on a 10-year launch program consisting of 10, 100, and 1,000 total launches.

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USAF COST OPTIMIZED BOOSTER SYSTEM STUDIES

(Contract No. AF 04(611)-7033)

22 September 1961

A. F. Program Structures Number 750 E
ARDC Project Number 3058
ARDC Task Number 79726

NORTH AMERICAN AVIATION, INC.
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FOREWORD

The work completed in this report, "Cost Optimized Booster System Studies", was accomplished under the sponsorship of Air Force Flight Test Center, Air Force Systems Command, United States Air Force.

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INTRODUCTION

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[redacted] covered by this report involved the investigation, analysis, and cost optimization of conventional two-stage tandem booster systems which utilized state-of-the-art and advanced concept propulsion systems in the 0.6 to 3 million pound thrust range. Numerous configurations utilizing various combinations of engine types, thrust ratings, and propellants were considered and optimized for both performance and costs. The most promising configurations in the 600 K and 1500 K thrust classes for both state-of-the-art and advanced propulsion systems were adopted for investigation of 800 K and 3000 K thrust booster systems.

Various alternate booster system concepts were also investigated and evaluated on a cost comparison basis with the selected two-stage booster system configurations incorporating advanced high chamber pressure engines.

The basis for evaluation and comparison of the various configurations and concepts was dollar cost per pound of payload in a 300-nautical-mile orbit. In order to determine economic trends influenced by varying launch rates, costing was based on a 10-year launch program consisting of 10, 100, and 1,000 total launches.

I. SUMMARY

SCOPE

The Cost Optimized Booster Systems Study is an investigation of booster systems with propulsive thrust in the 600K to 3000K class designed to inject payloads into a 300-mile circular orbit. In particular, it is a study of two-stage tandem single tankage cost optimized systems employing state-of-the-art and advance propulsion systems of 600K, 800K, 1500K and 3000K first-stage thrust for launch rates over a 10-year operational period of 10, 100, and 1 000. On a cost comparison basis with the conventional two-stage tandem configuration, several alternate concepts such as clustered first stage, single-stage-to-orbit, segmented solid first stage, lateral staging, LF_2/LH_2 second stage, paraglider recovery of a conventional first stage, and paraglider recovery of an extended first stage were evaluated. The basic parameter employed for comparison was dollars per pound of orbited payload.

OBJECTIVES

Objectives of this study were to determine cost optimum two-stage tandem booster system configuration, to determine optimum chamber pressure for the advanced high-pressure engine, and to identify ways to reduce R & D and production costs. Additional objectives of the study were the efforts to determine economic effects of recovery technique application to first-stage booster vehicles, and to evaluate the following alternate booster system concepts on a cost comparison basis with optimized tandem two-staged booster systems; (1) clustered first stage with optimized second stage, (2) clustered first stage with common module on both stages, (3) laterally staged booster systems, (4) conventional two-stage system with liquid fluorine/liquid hydrogen second stage, (5) conventional two-stage system with segmented solid first stage, and (6) single-stage-to-orbit booster system employing advanced high-pressure engine.

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First-stage thrust per module:	600 K, 800 K, 1500 K and 3000 K.
Engine types:	State-of-the-art—Conventional Bell Advanced —Plug and High Pressure
Propellants:	First stage - LO_2/LH_2, LO_2/RP-1, and Segmented solid; second stage-LO_2/LH_2 and LF_2/LH_2
Thrust to weight ratio:	First stage - 1. 2 Second stage - Optimized
Mission:	300-Nautical-Mile Orbit
Launch Rate:	10, 100 and 1000 over a 10-year operational period
Comparison Basis:	Dollars per pound of payload in orbit predicated upon calculated mission reliability

STUDY EMPHASIS

The emphasis was applied to selection studies of various propulsion systems for first stage of two-stage tandem booster configurations. These selection studies were confined to 600K and 1500K thrust. Alternate LO_2/LH_2 propellant booster system concepts were evaluated on the basis that advanced high-pressure engines are to be employed. Propulsion system studies were centered primarily on advanced high-pressure engines. The influence of reliability on costs is to be illustrated.

APPROACH

At the beginning of the study, a mechanization program was formulated for utilizing the IBM 7090 computer to aid in determining performance, payloads, weights, and production costs. This approach made it possible to consider far more parameters and configurations than would otherwise have been possible in the short time period. It also simplified the performance optimization process and proved to be invaluable in the determination of weights and production costs.

Each selected booster system configuration was carried through a preliminary analysis involving the overall design of the booster system,

which included propulsion, control features, and structural efficiencies. Results from the design analysis and the mechanized performance program were utilized in setting up the machine program for determining weights, payloads, and production costs. Final selection and alternate concept evaluation were based on the results of the cost analysis which included effects of overall booster system reliability.

DISCUSSION

PROPULSION

The engines selected for study were intended to represent a cross section of the most promising designs and concepts for vehicles of the thrust range specified in the contract (600 K to 3000 K). Propellants considered were LO_2/RP-1, LO_2/LH_2, LF_2/LH_2, and a representative solid propellant such as polyurethane or polybutadiene acrylic acid. The engines selected which are under active development are the F-1 and J-2. Engines not under active development at the desired thrust level were 150 K (SL) LO_2/LH_2, 800 K LO_2/LH_2 engine, a modified E-1 (600 K), a LO_2/LH_2 plug nozzle engine, a segmented solid engine, the high-pressure engine, and a LF_2/LH_2 engine. It was felt that some combination of engines and propellants in this group would represent a cost optimized boost vehicle from the propulsion system standpoint.

Inasmuch as the characteristics of state-of-the-art engines are well known, most of the engine system effort was concentrated on the evaluation of the Pratt & Whitney advanced high-chamber pressure engine.

High-combustion chamber pressure (P_c>1000 psia) provides several advantages for a pump-fed liquid rocket engine. Among these are good low-altitude performance, the capability of increasing high-altitude performance, small physical size and possible weight reduction over the more conventional engines. The increase in combustion chamber temperature due to the increase in chamber pressure does not appear to present a serious problem. It is predicted that conventional cooling methods, such as regenerative fuel flow and film cooling, will be satisfactory. Attitude control of a vehicle utilizing the subject engine may present more of a problem.

One of the first steps during the study was to determine the most desirable combustion chamber pressure for the vehicles to be studied. Estimated performance curves generated by Pratt & Whitney indicated that for optimum sea-level expansion a specific impulse increase of 7.4 percent would be realized by increasing chamber pressure from 1000

to 3000 psia, while only 2.8 percent increase would be obtained between 3000 and 5000 psia. Engine weight increases 29.2 percent when chamber pressure is increased from 1000 to 3000 psia, and a 57.2 percent increase accompanies a chamber pressure increase between 3000 and 5000 psia. The preceding comparison is presented for a thrust level of 600 K and considers bare engine weight (plus boost inducer) only. It is recognized that the gimbal actuator system and thrust mount weights would also increase. The effect of this weight increase was to decrease the booster mass fraction. Estimated development cost of the 5000 psia engine is 16.2 percent higher than that of the 3000 psia engine, while production costs are 7.83 percent higher. The payload comparison showed that, although the difference was small, the 3000 psia system provided the higher payload. As a result, the 3000 psia system was selected for further configuration study.

Evaluation of liquid propellant boosters were based on the use of the following oxidizer/fuel combinations: liquid oxygen-liquid hydrogen (LO_2/LH_2), liquid oxygen-RP-1 (LO_2/RP-1), and liquid flourine - liquid hydrogen (LF_2/LH_2).

No particular problems are anticipated with any of these combinations in respect to material compatibility. The corrosion resistance of all material of construction used with flourine depends on the formation of a passive flourine film. Therefore, the only additional requirement necessary to prepare hardware for service is passivating the material, which ordinarily would not apply to other cryogenic propellant hardware.

Design and installation of line and components necessary to handle the required quantity of propellants should present no unusual problems. Propellant loading systems associated with large boosters require auxiliary vent valves for LO_2, LF_2, and LH_2 during fill operations to allow for high initial propellant boil-off created by cool-down of propellant tanks during fill. In the case of multiple-tank modules, intertank propellant manifolding is desirable to allow all tanks to be filled simultaneously, thereby eliminating intertank temperature differential problems.

To satisfy turbopump operating requirements, it is necessary to provide some means of supplying the propellants under pressure to the pump inlets. Pump inlet pressure requirements are a function of the turbopump design and vary with the individual engines. Two common methods of supplying the propellants under the necessary pressure to the turbopump inlets are by a boost pump system or by ullage pressure only.

The use of boost pumps to supply pump inlet pressure requirements is attractive because they reduce tank pressurization, reduce line insulation

needed to maintain NPSH requirements at end boost, reduce problems of loss of turbo NPSH from heat leak, and may permit use of a high-pressure turbopump of higher speed and less weight.

It is recognized that the use of boost pumps does not delete the pressurization system since it is desirable to maintain propellant vapor pressure; however, the weight of the system is decreased and in some cases the weight decrease can be considerable. Tank structural weights will be reduced, but the decrease in weight is not a direct function of ullage pressure since the tank skin thickness is also a function of longitudinal loads as well as hoop tension. In no case can the ullage pressure be lower than the ambient due to the weight penalty suffered by any propellant tank compressive loads. Structural weight savings, then, are dependent on tank geometry to a considerable degree.

Several methods have been used for pressurizing propellant tank ullage pressures. In the course of the study five basic system concepts were evaluated. These were: solid grain pressurization, high-pressure gas storage, low-pressure storage (liquid pressurant) with vaporization, products of combustion, and propellant vaporization. In comparing these systems, emphasis was directed toward the high-pressure engine with the thought that more flexibility can be assumed in regard to the engine's role in support of the pressurizing system. Vaporized rocket engine propellants for pressurizing tank ullage results in one of the most desirable pressurization systems. The reliability of a propellant evaporative system is high because of the elimination of a separate storage container and its accompanying controls. LO_2 and LH_2 propellants are readily adapted to an evaporative system. In a LO_2/LH_2 booster the use of hydrogen on the liquid hydrogen presents the optimum system. The evaporation of LO_2 for oxidizer tank pressurization is attractive, and although not the lightest, it represents the optimum system. RP-1 fuel does not lend itself to an evaporative system and liquid fluorine, although adapted to an evaporative system, is not recommended because of its high molecular weight.

FLIGHT MECHANICS

In the course of this study program, primary emphasis was placed on a detailed parametric performance analysis of the booster systems. An IBM 7090 booster design program has been developed for estimating the performance of multistaged boost vehicles for various earth, lunar, and space missions. The program includes the effects of vehicle drag, gravity losses, rotating earth, and the variations in thrust, specific impulse, and gravitational attraction with altitude. The variation in booster mass fraction with booster size and thrust level is also taken into account.

In order to minimize the machine time required for large parametric performance studies, the program was based on a ballistic boost path for the two stages. A ballistic path does not represent the optimum path and will not result in the maximum payload capability of a given boost system. However, past studies have shown that the ballistic path will result in near maximum orbited payload as long as the second-stage initial thrust-to-weight ratio, $(T/W_o)_2$, is greater than 1. For example, an optimum trajectory calculation for a typical Saturn configuration where $(T/W_o)_2$ was approximately 1.6, yielded an increase in payload capability of less than 3 percent over a pure ballistic path. As a consequence, the payload trends and magnitudes based on a ballistic path should be valid for this study since in the majority of the cases $(T/W_o)_2$ is greater than 1. For the particular mission studied, namely, a two-stage vehicle boosted into a 300-nautical-mile circular orbit, a variety of operational modes can be achieved. For this study the orbital mission was achieved assuming 180-degree coast to apogee following the thrust cut-off with a kick-into-apogee to impart the orbital velocity. It may be that this mission profile is undesirable, but the assumptions so made permitted straightforward computations with realistic results.

The subjects treated under dynamic considerations were separation, vehicle bending, propellant sloshing, and staging. The information presented in this section was excerpted from recent S&ID studies on large booster systems in the thrust range of interest. This has been done because detailed analyses of dynamic considerations for the myriad configurations discussed in this report were clearly impossible within the established funding, and because the work previously completed on dynamic considerations for large booster systems was representative in nature and perfectly valid for the configurations investigated in this report.

Performance analysis results indicated that when RP fuel was used in the lower stages and LH_2 fuel used in the upper stages, staging optimization favored the LH_2 stages for propellant mass fractions. The optimum ΔV_1 for the 600 K systems ranged from 4000 to 8000 fps whereas for an all LO_2-LH_2 (lower stage), LO_2-LH_2 (upper stage) system the ΔV_1 ranged from 8000 to 14,000 fps. Predicated upon the type of second-stage engine, the optimum performance $(T/W_o)_2$ for the P & W high-pressure concept was 2.15 and for the J-2 concept was 1.57. To arrive at a selected system's true optimization, an iteration procedure is involved inasmuch as the engineering analyses involving propulsion, structures, system weights, and guidance are initially conducted separately. A very close coupling exists between the various analyses whereby a change in structure weight will affect the propellant weight, which in turn changes the payload weight, which changes the structure weight. This coupling arises because the outputs of each analysis serve as inputs for the others.

For example, the results of performing two iterations increased the payload weight from an initial maximum of 32, 500 pounds to a final maximum of 40, 500 pounds.

VEHICLE

The vehicle configurations have been generated to support the booster systems cost optimization program by confirming the validity of the booster arrangements and assumptions, also to delineate problem areas which are associated with specific types of engines, propellants, or systems. The designs presented for the machine program are the result of a single iteration in most cases; that is, after the thrust level was established and the mass ratios were estimated from the curves established by the weights group, a configuration was developed. In all configurations the fuel is located in the upper tank while thc oxidizer is placed at the low level. The engine mounts have been standardized by using a single cone or double cone, as the situation might require. For cases where more than three engines were used, the standard engine mount ring has been selected as the design upon which to base the booster weights. It was assumed that actuated arms on the ground equipment could attach to the bottom of the tank to support the vehicle while it is on the launch pad and thus impose no scar weight on the booster itself.

The design effort served as common ground for all technical groups. The resulting effort fell into three categories: (1) single tank two-stage booster systems, (2) alternate booster systems, and (3) engine installation studies. Two-stage booster systems studies placed emphasis on configurations with thrust levels of 600 K and 1500 K when it became apparent that the possible number of configurations would be too large to handle and still stay with the scope of the study. The 800 K and 3000 K systems would be predicated upon the cost optimized 600 K and 1500 K systems respectively for the purpose of providing proper costing data throughout the thrust range specified. Later in the study it became apparent that within a thrust class no one system could be identified for the range of launch rates. Therefore, it was decided that the most promising booster system employing the state-of-the-art propulsion systems and advanced propulsion systems would be studied as selected systems. Selected advanced booster systems employed 3000 psi high-pressure propulsion systems and are illustrated in Figure 1. Selected state-of-the-art booster systems employed J-2 and F-1 propulsion systems and are illustrated in Figure 2. Further details are presented in the analysis section of this report.

Alternate concepts studied are presented in Figure 3 and are as follows: (1) 600 K first-stage clustered modules with common second-stage module and cost optimized second stage, (2) paraglider recovery-for 600 K and 1500 K conventional first stage and 600 K extended first stage,

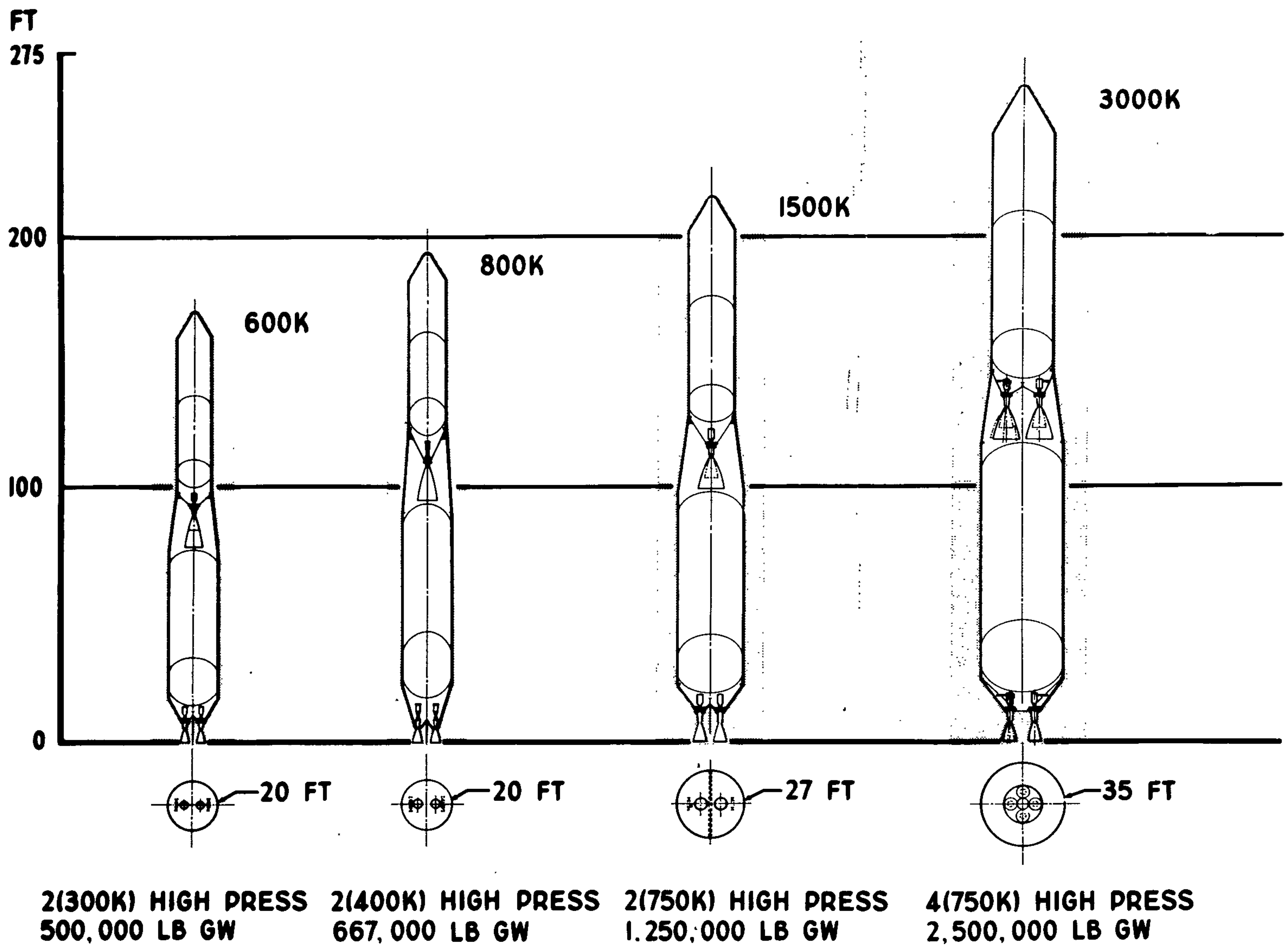

Figure 1. Advanced Booster Systems

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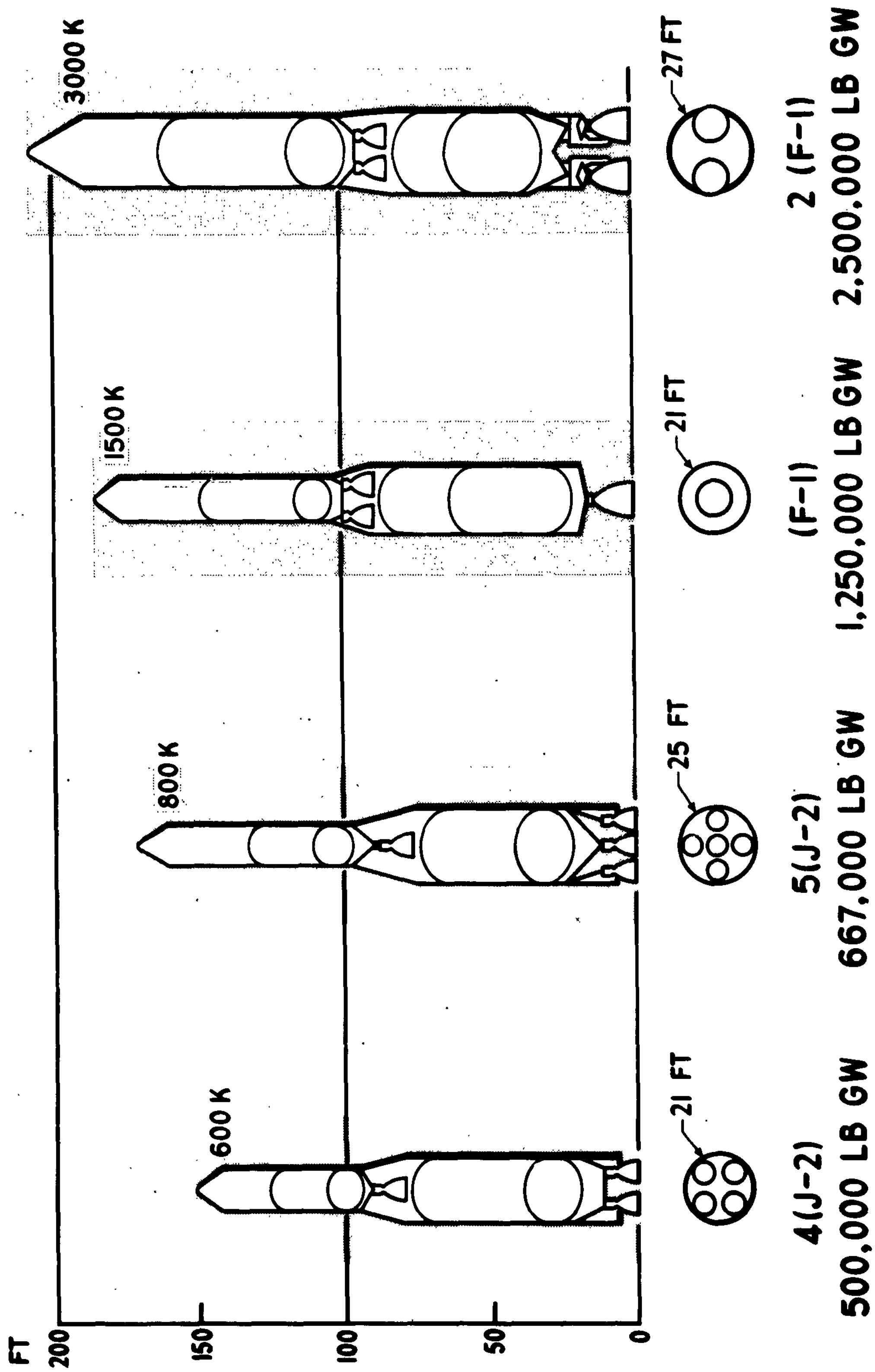

Figure 2. State-Of-The-Art Booster Systems

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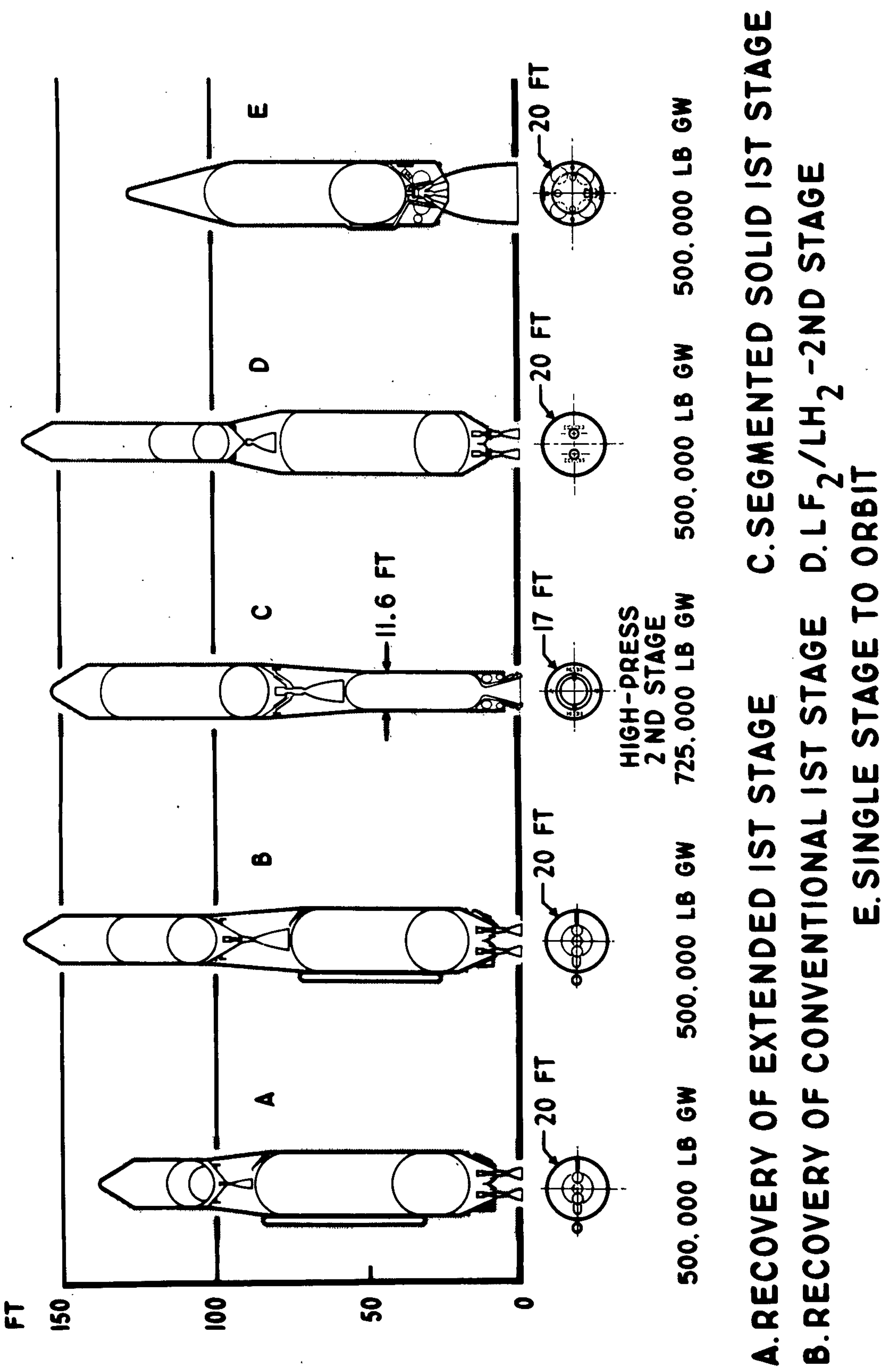

Figure 3. 600K Alternate Booster Concepts

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(3) 600 K segmented solid first stage with selected cost optimized second stage, (4) 200 K LF_2/LH_2 cost optimized second stage on the selected 600K LO_2/LH_2 first stage, (5) 1500K lateral stage system included Douglas Aircraft's common tankage concept and North American Aviation, Inc. common-engine concept, and (6) 600 K single-stage-to-orbit employing the advanced high-pressure engine with a secondary expansion nozzle. Most of the alternate concepts were not applicable to the performance-weight-cost mechanization study; therefore, the program was used only to establish weight of their peculiar design variations.

Engine installation studies were primarily concerned with the advanced high-pressure engine. Earlier studies performed at S&ID served as the basis of the J-2 and F-1 engine installation. The optimum-thrust structure was also based on these studies. One of the greatest problems which occurred while studying the high-pressure Pratt & Whitney engine was the method whereby this engine could be gimbaled without degrading the reliability of the main propellant feed lines. Since the engine is designed with the intake in the forward end, it was necessary to show methods where the flexing of these intake lines could be held to a minimum during gimbaling conditions. Two methods of gimbaling the high-pressure engine were studied. The first method places a gimbal box similar to that used on the J-2 at the forward end of the engine. This method has the advantages of producing less propellant line length and less flexure than the second method studied. The second method places a gimbal point at the forward end of the thrust chamber which is a point approximately 1/3 of the distance from the forward end of the engine to the nozzle exit. This method results in longer propellant lines and slightly more line flexure than the first method and is the approach employed in this study for the following reasons: (1) excursion of the nozzle exit for a given deflection angle is 1/3 less than for the first method; (2) the distance from the aft end of the LO_2 tank to the nozzle exit is about 5 feet less for a 300 K engine or an overall length of 11 feet in a two-stage system; (3) the method of gimbaling at the forward end will produce a high bending moment in the turbopump housing (high moment of inertia) which will not occur if the engine is gimbaled near the center; and (4) the extension of the forward portion of the engine into the thrust structure as in the second method produces an ideal location for gimbaling actuators and permits a wide selection of moment arms for the actuator attach point. The thrust structure also acts as the flame shield.

STRUCTURES

The purpose of the structural analysis was to provide a sound basis for an accurate determination of the structural weight and costs involved in the various configurations under study. Analysis was made in sufficient detail to establish the validity of basic load paths and structural components.

Because of the vast number of configurations under study, a machine program has been utilized to calculate the weight and cost of all the basic structural components. However, hand calculated "base points" are required as machine inputs. In the initial phase of the study, base points established in the previous study were employed. Booster tankage was of skin-stringer construction with ullage pressures of 35 psia and 50 psia respectively for the LH_2 and LOX tanks. In the later stages of this study, specific configurations were used for detail stress analysis and subsequent use as base points for the machine program to determine its influence on systems cost. In this case the booster tankage was designed as sandwich construction and tank ullage pressures were reduced to the minimum needed to meet NPSH requirements of the pumps. The new ullage pressures are approximately 30 psia in both LH_2 and LOX tanks for a typical arrangement.

The tank walls are designed to be free standing while unpressurized during the prelaunch phase and for body bending and axial loads encountered during flight. A monocoque sandwich construction was found to have a small weight advantage over conventional skin-stringer construction for the tank walls. The skin gages were set by hoop tension requirements resulting from internal pressure with sufficient core material being added to stabilize the facing sheets under axial compressive loads.

The forward and aft headers are subjected to bursting pressure only and are therefore of membrane construction in an elliptical shape to minimize weight. In configurations where a portion of the aft header is incorporated as part of the thrust structure, these segments are subjected to compressive loads and are of sandwich construction.

The aft header is generally critical at end boost when it is subjected to ullage pressure plus dynamic head. The center divider is subjected to both bursting and collapsing pressures. Here an ellipsoidal sandwich construction produced the minimum weight.

The function of the thrust structure is to transmit the concentrated thrust of the engines to the booster outer shell as uniformly and as directly as possible with a minimum weight structure. For a single engine located at the tank centerline, a conical thrust structure is ideally suited. A multiple-engine cluster required tapered integral thrust longerons to accept the concentrated thrust at the engine pickup points, a conical frustum to distribute the concentrated shear loads uniformly to the cylindrical tank, and a frame at each end of the conical segment to realign the thrust loads.

A method was devised for a two-engine thrust structure whereby two intersecting cones were employed. The design requires a fitting at the apex of each cone, a parabolic arch at the intersection of the two cones, two integral longerons at the junction points of the tank and arch, and a frame at the base of the dual cones.

WEIGHTS

Inasmuch as the study involved numerous configurations and parameters, it was necessary to develop a two-phase mechanization program utilizing the IBM 7090 computer to expedite the generation of required data. Phase I was required for initial performance factors and Phase II established structural, propellant, and payload weights.

Initially it was necessary to generate estimated propellant mass fractions for the various configurations studied. Data was available from the previous study for the first iteration of the 1500K thrust but had to be developed for the 600K thrust systems. As the configurations became firm, it was possible to determine more accurate weights.

Input variables relative to engine weight and thrust, number of engines, propellant mass fractions, and propellant unit weight were programmed into the computer. The output consisted of a complete breakdown of payload, vehicle stage, and propellant weights for each configuration considered.

RELIABILITY

It was realized that in order to establish systems cost to a reasonable degree of accuracy, a fairly comprehensive reliability study was required. Results showed that a highly successful vehicle program can be attained through the development of boosters based upon simplification of systems and utilization of equipment which has demonstrated high reliability. The use of equipment and materials within the state of the art for the program will reduce many of the uncertainties associated with development programs. Until such time as large solid rocket motors demonstrate reliabilities at the levels of confidence attainable with liquid propellant boosters, it seems advisable to include them in the early phases of an operational program.

The study has also disclosed the differences in reliability resulting from two program funding philosophies. The most common of these is the one praticed in many of our current missile programs. It involves a very conservative appropriation of monies during the R & D program. The consequences of such an action are that inadequate emphasis is placed on the importance of reliable designs. Vehicles are launched without sufficient knowledge of the reliability of many of the components, sub-systems, and systems. Low reliability is the product of this funding philosophy. The cost of supporting such a program (because of low reliability) and improving reliability from data obtained from flight failures, becomes staggering. The more desirable funding program provides sufficient R & D funds to design highly reliable components and systems, and conduct tests which demonstrate high reliability prior to design freeze.

Adequate testing in the predicted environments (including combined environments) is required to substantiate the value of attained reliability at a high confidence. The result of this type of program is high initial launch reliability and low operational support because of the reduction in mission failures.

The savings realized in reduction of vehicle losses over the normal funded program provides the necessary additional R & D funds. As a result, the total cost of an accelerated R & D funded program will be no more than that of the normal funding program and in this study has been predicted to result in a lower total program cost.

The analysis of engine manufacturer's data revealed that presently developed liquid-propellant engines (F-1 and J-2) offer reliability advantages over a high-pressure liquid system. The solid-propellant motor reliability data predicted by the manufacturers do not reflect the values demonstrated by current large rocket motors. However, to provide a complete estimate, the manufacturer's reliability predictions and the values presently demonstrated by large solids were used in the vehicle comparisons.

Vehicles utilizing the F-1 and J-2 engines are predicted to have the highest reliabilities. This conclusion is formed without considering the use of an engine-out capability in the configuration with a cluster of four J-2's in the second stage.

The laterally staged configuration developed by Douglas Aircraft presents reliability problems because of the size and number of propellant line disconnects. It was not possible to compare the reliabilities of this vehicle with the tandem staged vehicles because the prediction contained in the Douglas report was without reference to a specific calendar time.

The laterally staged booster designed by S&ID eliminates the line disconnects between stages and reduces the separation problems significantly. A more thorough analysis of this configuration is required to predict its reliability.

ECONOMICS AND OPERATION

A comprehensive computing machine program covering performance and weights analysis was extended to include the calculation of production costs. Certain quantities which were obtained from the weights program were readily useful in the cost calculations. Among these were the AMPR weight, the type and quantity of engines, the propellant weight, and the payload weight. From the vehicle description and AMPR weight, the fabrication and tooling costs were estimated; from the fuel and oxidizer description and weight the propellant costs were estimated. In a similar manner the engines

were costed using vendor data where available. A calculation procedure was adopted which permitted the evaluation of both modular and recoverable alternatives as well as the conventional single-tank designs. By use of this procedure, the lowest production cost per pound of payload could be readily obtained and compared for each configuration and the most promising determined. It was not assumed that the highest payload configuration would necessarily be the most attractive.

The economic analysis of the study configurations was directed toward arriving at a booster dollars-per-pound-of-payload as a common comparison for each configuration. The expected results were to determine the least expensive or cost optimized configuration in terms of total program cost per pound of payload boosted to a 300-nautical-mile circular orbit. The total program cost components of each configuration, consisting of R & D, production, and operations costs, were considered on the basis of thrust level for the four configuration groups — 0.6, 0.8, 1.5, and 3.0 million pounds of thrust. These costs cover a 3-year development and 10-year operational launch period and have been reduced to a booster cost-per-pound-of-payload as a basis for configuration comparison. The operational launches occur in the 1965 to 1975 period.

The costs are compared both as to the minimum total program costs and the minimum cost per pound of payload. These cost parameters were studied over the prescribed launch rates to determine any crossover or inflection points. The meaning of such points were examined as to their effect on ultimate vehicle selection.

The first phase of the study covered primarily selected two-stage vehicles in the lower thrust range. This part of the study shows the desirability and feasibility of developing two-stage vehicles for high-payload orbital missions by 1965. Boosters with low dollars-per-pound-of-payload costs can be available within the next 3 years using combinations of existing state-of-the-art J-2 and F-1 engines in the first and second stages or within 4 years if advanced high-pressure engines are used in both first and second stages. The second phase of the study covered comparison of alternate systems with the selected two-stage systems.

The most promising two-stage vehicles selected for further analysis in the study fall into two booster configuration groups, (1) boosters using state-of-the-art engines such as J-2's and F-1's and (2) boosters using advanced high-pressure LO_2/LH_2 engines.

The trade-off curves showing dollars per pound of payload versus launch rates for the four thrust levels of 0.6, 0.8, 1.5, and 3.0 million pounds of thrust show that the cost per pound of payload decreases with increasing total thrust level for any given engine and propellant combination,

with the greatest relative decrease occurring at the lower thrust levels of 0.6 and 0.8 million pounds of thrust. In comparing the state-of-the-art systems with the advanced high-pressure systems, it can be noted that there is no significant difference between the two within each of thrust classes and launch rates covered. The systems utilizing existing state-of-the-art systems remain competitive with systems utilizing advanced high-pressure engines due to the reduction in R & D costs and higher initial reliability. If the full R & D costs for the state-of-the-art propulsion system were included, the configurations utilizing advanced high-pressure engines would show a distinct advantage.

The ratios between the three major cost elements (R & D, production, and operations) vary with the launch rate when considered on a cost-per-pound basis. The higher the launch rate, the more significant the production costs per pound of payload becomes. At 10 launches per 10-year period, the R & D and operations costs are 85 percent of the total with production amounting to only 15 percent. At 100 launches per 10-year period, the R & D, production and operations-costs-per-pound-of-payload are all approximately equal. At the higher rate of 1,000 launches for the same period, the production costs per pound of payload constitutes about 80 percent of the total costs. These ratios were typical for all the two-stage system studied.

When the costs for each configuration were tabulated, it was revealed that the higher the first stage thrust level, the cheaper the dollars-per-pound-of-payload for any given launch rate. However, it should be noted that if the dollars-per-pound-of-payload orbited is compared on the basis of total successful pounds orbited rather than launch rate, the reverse is true. This is a result of the smaller system orbiting payload earlier at a higher launch rate than the larger system, coupled with reduced prices through high production. If an extremely large amount of payload is orbited, the larger system will eventually show an advantage.

All development and production cost results presented are based upon a calculated launch reliability of less than 100 percent. Overall booster costs-per-pound-of-payload in orbit were determined based on an estimated mission reliability for each configuration.

The range of reliabilities for boosters covered in this study run from 80 through 96 percent. The effect of decreased reliability is to reduce the total payload successfully launched at any given launch rate and to raise the total cost per-pound-of-payload. These effects are the same for all configurations presented in this study.

The second phase of the booster study covered comparisons of alternate configurations with the selected two-stage systems. Among the alternate configurations considered were clustered stages, lateral staging, segmented

solid first stages, liquid fluorine second stage, paraglider first stage recovery, single-stage-to-orbit configurations, and extended first-stage configurations.

The concept of clustering modular tanks in the first stage to obtain boosters with higher thrust levels shows the best cost-per-pound-of-payload advantage at lower launch rates and, at the same time, has the advantage of greater program flexibility. The greatest economic advantage is noted if an advanced high-pressure propulsion system is employed.

The cost differences of the clustered vehicles over the reference vehicle are slightly more pronounced with the common-module configurations than with the cost optimized second-stage configurations. These two-stage clustered vehicles utilize a common module in both stages. Use of advanced high-pressure engines allows efficient expansion ratios for both stages, as opposed to the normal impulse reduction in conventional bell propulsion systems.

The laterally staged booster configurations are limited in their payload capability but are economically competitive with conventional two-stage booster systems. This advantage is most significant at the lower launch rates of 50 per 10-year period and is not reflected at the higher total launches. The reason for the economic advantage of the laterally staged configurations is that they are designed as semimodular and employ either identical tanks or engines.

The recoverable two-stage configuration starts to show a slight cost advantage in dollars-per-pound-of-payload at the 50 per 10-year period launch rate when compared with the expendable 1.5 million-pound thrust two-stage configuration. When the recoverable two-stage configuration is compared with the laterally staged configurations, the breakeven point occurs at the 100 per 10-year period launch rate.

The dollars-per-pound-of-payload curves of the two-stage 0.6 million-pound thrust vehicles indicate the expendable and recoverable systems at 100 per 10-year period are approximately equal in cost. The R & D and production costs of the paraglider recovery system tend to offset the savings in vehicle production costs from re-use of the same booster vehicle. Also, there is a payload penalty associated with the recovery system. At the higher launch rates, there is a definite program cost savings when the first stage is recovered.

The extended first-stage vehicle system which utilizes paraglider recovery is also compared with the expendable 0.6 million-pound thrust vehicle. Results show that extension of the first stage (higher first-stage increment velocity) with paraglider recovery is actually higher in cost over

the conventional stage recovery concept. This results from the payload capability degrading faster than incremental first-stage savings.

One of the alternate two-stage configurations studied was a LF_2/LH_2 second stage propulsion system on a cost optimized LO_2/LH_2 first stage system. This configuration showed no economic or payload advantages over the selected systems.

The study of a segmented solid first stage coupled with the cost optimized LO_2/LH_2 second stage showed no economic advantage for boosting comparable payloads into orbit. The solid propellant propulsion costs were determined from a composite of several vendor costs with a low, optimistic cost used where possible. All of the configurations covered in the study were considerably more economical than that utilizing a solid propellant first stage.

A review of the single-stage-to-orbit configurations indicates that their dollar per pound of payload are higher than the other 0.6 million-pound thrust vehicles. This is primarily due to a low payload capability and the requirement for a secondary expansion nozzle for the advanced high-pressure engines used.

II. CONCLUSIONS AND RECOMMENDATIONS

CONCLUSIONS

No optimum two-stage booster system emerged as an obvious choice within the range of thrust levels and launch rates investigated during the study. Although the advanced engines possess greater potential, the state-of-the-art engines are presently competitive as a result of higher initial reliability, cheaper propellant and systems cost in the case of RP-1 systems, and the fact that R & D costs have been previously absorbed.

The study revealed a definite correlation between performance and costs based on dollars-per-pound-of-payload in orbit. Results of comparing booster systems utilizing selected engine types of varying number and thrust show that performance optimized systems will generally reflect the lowest cost-per-pound-of-payload. This appears obvious when considered from the point of view that total program costs vary slightly within a given thrust range, whereas payload capability increases considerably with performance optimization.

Results of the economics and operation analysis revealed an interesting aspect of booster systems utilized for orbiting an unspecified payload weight. For any given launch rate, the cost-per-pound-of-payload in orbit decreases with an increase in thrust, whereas the opposite is true when dollars-per-pound-of-payload versus total payload is considered. In this case the cost-per-pound-of-payload in orbit decreases with thrust level. The significance of this apparent inconsistency is that savings derived from mass production and early amortization of development costs are realized much earlier with a smaller booster system which requires a higher launch rate to orbit an equivalent payload. If an extremely large amount of payload is orbited, the higher-thrust booster system will eventually show an economic advantage. This observation substantiates the well understood desirability of optimizing the booster system design for a specific payload mission.

Results of the study indicate that 3000 psi is the optimum chamber pressure for the advanced type Pratt & Whitney engine. Although an increase in chamber pressure will result in higher specific impulse at low altitude and lend itself to an increase in area ratio for optimum expansion, the associated weight penalty and increased development and production costs nullify these advantages at pressures above 3000 psi.

Inasmuch as an optimum cost optimized system could not be identified, it was difficult to identify specific ways to reduce R & D and production costs of future propulsion systems. However, it appears that the most promising area for concentrating effort towards this end would be to develop a single optimized high-pressure LO_2/LH_2 engine for both first- and second-stage application. This approach would culminate in a versatile engine of high-performance characteristics that could be utilized on either tandem, single-stage, or clustered-modular configurations.

The paraglider recovery technique application on the first stage of a conventional tandem-staged booster system appears to be the most economical booster system approach for high launch rates in excess of 10 per year over a 10-year period. No economic advantage is gained by recovery of an extended first-stage version of the two-stage booster system.

The clustered configuration utilizing common modules on both stages won out as the most economical booster system at the low to medium launch rates of up to 10 per year over a 10-year period. At higher launch rates, this system gives way to the paraglider recovered first-stage booster on the basis of cost-per-pound-of-payload in orbit.

No economic advantage is gained by optimizing the second stage of a clustered configuration. Study results show this scheme is more economical at low launch rates than the conventional tandem-staged system, but not as good as the clustered configuration utilizing a common module on both stages.

Laterally staged boosters are economically competitive with conventional two-stage booster systems. It appears to have a somewhat lower cost below a launch rate of about 5 per year over the 10-year period. The S&ID concept is a semimodular design; whereby either the propulsion system or tankage may be truly modular with consequent savings derived from reduced R & D and production costs.

The liquid flourine-liquid hydrogen propulsion system for second-stage application does not provide an economic or payload advantage over the advanced high chamber-pressure engines, or an economic advantage over state-of-the-art systems.

The segmented solid first stage appeared economically unattractive below approximately 100 launches per year over a 10-year period compared to the liquid propellant first stages.

The single-stage-to-orbit concept is feasible predicated upon employment of a secondary expansion nozzle for the advanced high-pressure engine. This concept does not lend itself to gimbaling and must be provided with a

separate flight control system. Preliminary analysis indicated a payload capability of between 20,000 and 30,000 pounds, compared to 40,000 pounds for the selected conventional two-stage system.

RECOMMENDATIONS

The development of the high-pressure engine cycle currently proposed by the Pratt & Whitney Aircraft Company should be continued. This type of engine has been shown to have advantages when used as a common propulsion unit for both first-and upper-stage applications, particularly in the various modular concepts. In addition, the high-pressure engine cycle shows significant performance gains which have not been fully exploited in this study. For example, this engine makes the single-stage-to-orbit concept far more promising than in the past with other chemical systems.

This recommendation is further substantiated by the results of earlier advanced booster studies conducted for the USAF which showed that economic and performing advantages were obtainable with advanced engine systems such as the plug nozzle. It is possible that research and development effort directed toward the combination of desirable features of the high-pressure engine and some of the plug nozzle concepts may prove rewarding.

Research and development activity should be continued with the modular concept. Both economic gains and wide-range growth potential have been shown for various versions of this concept. Both first and upper stages should be considered with the clustered or modular capability.

First-stage recovery utilizing the paraglider concept shows economic advantages at launch rates above 10 per year. Cycle incorporation of a recovery concept such as the paraglider is contingent upon USAF mission, operational, and launch site considerations. Continued research and development with the paraglider recovery system is recommended.

The single-stage-to-orbit launch vehicle systems were not shown to have economic advantages. However, the systems considered for such a mission utilizing the Pratt & Whitney high-pressure engine cycle with a secondary expansion nozzle showed substantial payload capabilities and potential economic and payload improvements. Future USAF requirements and advances in the structures and propulsion state of the art may make the single-stage, orbital system extremely attractive. Systems such as those covered in this report may be competitive with the aerospace plane concept. As suggested above, research and development activities in the high-pressure engine as well as in light-weight tankage in subsystems areas should be continued leading toward the development of a single-stage orbital rocket system.

III. STUDY PARAMETERS

SCOPE

The Cost Optimized Booster Study is a follow-on of USAF Research and Development Study of Advanced Propulsion Systems, AF 04(611)-6069, which was primarily a feasibility and economics study conducted to determine various characteristics and requirements of optimum first-stage boosters in the 1.5 to 20 million-pound thrust range. The Cost Optimized Booster Study confines the first-stage thrust to between 0.6 and 3 million pounds and extends the initial study to include optimization of both stages of a conventional two-stage booster system. The study also includes the optimization and evaluation of various alternate booster system concepts, consideration of recovery technique application to first-stage vehicles and further investigation of state-of-the-art and advanced type propulsion systems, especially the Pratt & Whitney high chamber-pressure engine. Evaluations, comparisons, and cost analysis are based on dollars-per-pound-of-payload placed in a 300-nautical-mile orbit.

CRITERIA

Inasmuch as this study is a continuation of the previous study program, AF 04(611)-6069, which indicated the merit of further study activities into various areas, a review of the conclusions and recommendations resulting from that study will reveal the basis for formulation of the following objectives, ground rules, and study emphasis established through negotiations and liaison between the USAF and S&ID.

OBJECTIVES

The major study objectives are the determination of:

1. An optimum two-stage single-tank booster system configuration, within limits of study ground rules, that may be utilized for boosting a payload into a 300-nautical-mile orbit.

2. An optimum chamber pressure for the advanced high-pressure liquid rocket engine.

3. Specific areas of state-of-the-art propulsion systems where applied research can reduce development and production costs of future propulsion systems.

4. Specific ways to reduce R & D and production costs of the most promising booster systems.

5. Economic effects resulting from the utilization of recent recovery techniques on laterally staged vehicles and the first-stage vehicle of the optimized two-stage booster system.

6. Economic gains to be realized from a two-stage booster system design consisting of a small expendable second stage and a recoverable first-stage vehicle that utilizes advanced high-pressure engines and accomplishes the greater portion of boost.

In addition to the above objectives, the study encompasses the cost optimization of the following alternate booster system concepts which are evaluated and compared on a cost basis with the optimum conventional two-stage booster system:

1. Clustered first stage with single second stage utilizing common modules on both stages.

2. Clustered first stage with cost optimized second stage.

3. Laterally staged booster systems.

4. Conventional two-stage booster system utilizing a LF_2/LH_2 second stage.

5. Conventional two-stage booster system utilizing a segmented solid first stage.

6. Single-stage-to-orbit booster system employing the advanced high-pressure engine.

GROUND RULES

The basic ground rules established in USAF Research and Development Study of Advanced Propulsion Systems, AF 04(611)-6069, apply to this study except where modified by the following specific ground rules which, in the main, define the limitations and guiding parameters that were necessary to conduct the study and channel the effort toward attainment of study objectives:

1. First-stage thrust level is limited to 600,000; 800,000; 1,500,000; and 3,000,000 pounds per module.

2. Engine considerations are confined to state-of-the-art conventional bell engines and advanced engines of the plug and high chamber-pressure types.

3. Conventional engine chamber pressure is in the range of 800 to 1000 psi for first-stage application. Second-stage engine chamber pressure is a function of vehicle cost optimization.

4. All second-stage vehicles except the LF_2/LH_2 configuration use LO_2/LH_2 propellants. Both LO_2/RP-1 and LO_2/LH_2 are considered for first-stage vehicles.

5. Thrust-to-weight ratio for all first-stage vehicles except the solid and module configuration is confined to 1.2 and optimized for the second stage.

6. The booster system mission is to inject a payload into a 300-nautical-mile circular orbit.

7. All booster systems are cost optimized.

8. Cost comparisons are based on dollars per pound of payload in orbit.

9. Costing is based on a 10-year launch program consisting of 10, 100, and 1000 total launches.

STUDY EMPHASIS

During the early phases of the program, it became apparent that to keep the effort within a reasonable scope it would be necessary to concentrate the program costs, weights, and performance mechanization studies on the 600K and 1500K conventional two-stage booster systems. After the optimum configurations were determined at these thrust levels for both state-of-the-art and advanced engines, similar propulsion systems would be utilized for optimization of the 800K and 3000K configurations. Alternate booster systems, except the liquid fluorine second-stage and segmented solid first-stage vehicles, would be optimized and evaluated on the basis of using advanced type engines.

APPROACH

The cost optimized booster study covered in this report was conducted during a 4-month period beginning on 1 May 1961.

At the beginning of the study, a mechanization program was formulated for utilizing the IBM 7090 computor to aid in determining performance, payloads, weights, and production costs. This approach made it possible to consider far more parameters and configurations than would otherwise have been possible in the short time period allotted in the contract. It also simplified the performance optimization process and proved to be invaluable in the determination of weights and production costs.

The study program was organized generally in accordance with the block diagram shown in Figure 4 to provide an orderly sequence of the accomplishments of study objectives. Many individual portions of the study were conducted concurrently and involved initial assumptions and iterative processes for the correlation of design, weights, costing, and other data.

Basically, the program effort involved a preliminary concept definition of each booster system configuration, including the propulsion system and assignment of major parameters. These parameters involved estimated stage mass distribution, estimated structural efficiencies, and others from which preliminary size and performance characteristics were established. With the estimates of structural criteria, aerodynamic characteristics, engine performance, propulsion subsystem characteristics and other material, the mechanized performance program was employed. Comparison of results with initial estimates and further iteration resulted in performance optimized systems.

Each selected booster system configuration was carried through a preliminary analysis involving the overall design of the booster system which included propulsion subsystems, control features, and structural efficiencies. Results from the design analysis and the mechanized performance program were utilized in setting up the mechanized program from determining weights, payloads, and production costs. Final selection and alternate concept evaluation were based on the results of the cost analysis which included effects of overall booster system reliability.

It should be noted that for an economic analysis such as conducted for this study where total program costs are considered, the magnitude of the

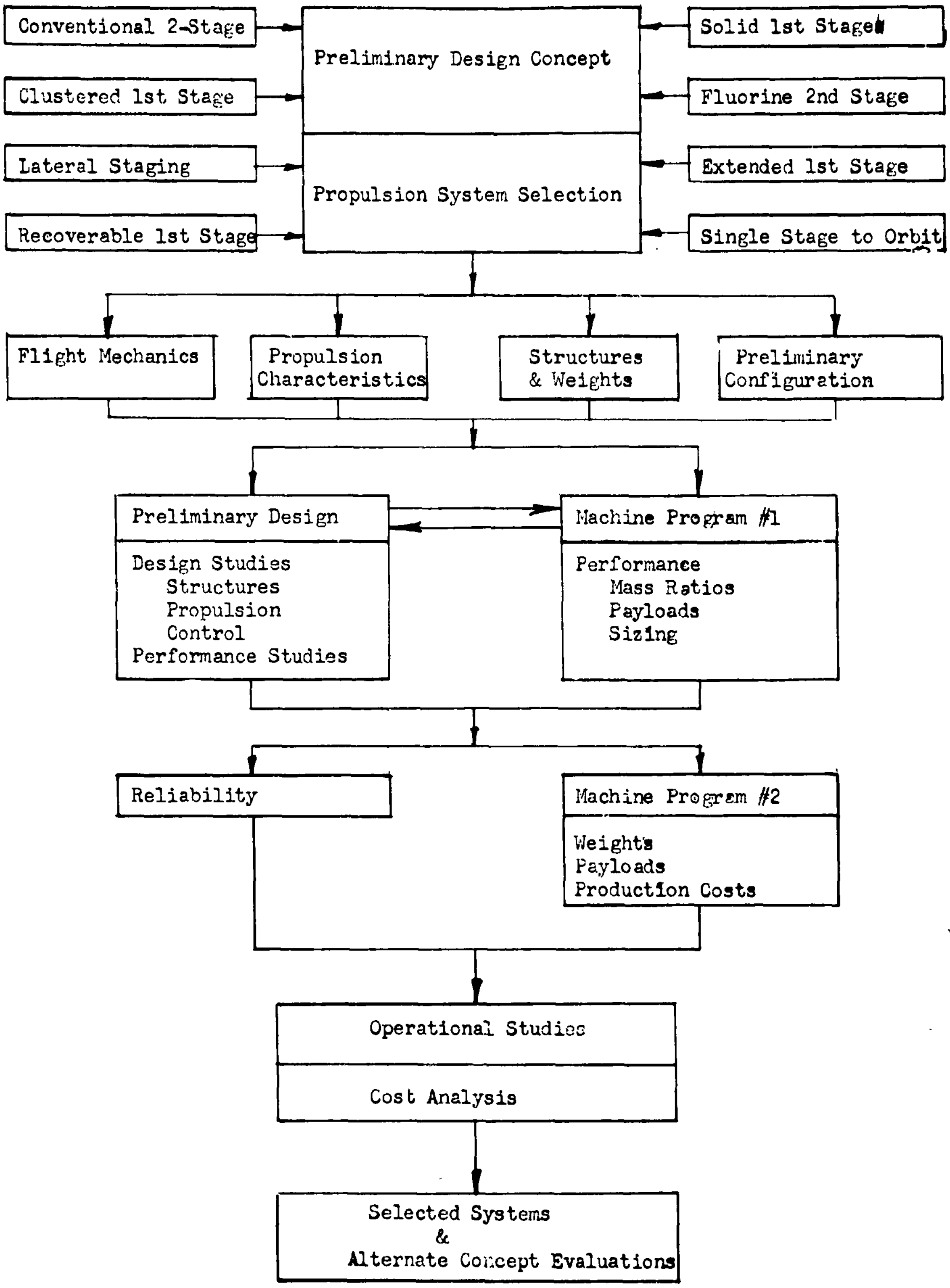


Figure 4. Cost Optimized Booster Systems Study Program

fixed program costs tend to make the cost differential between the various systems relatively small. Reliability and payload capability have a pronounced effect on costs in terms of dollars per pound of payload in orbit, although total program costs are otherwise equal. For this reason, the question of reliability was given considerable attention during this study and results were applied in the cost analysis area to determine the relative cost of various configurations.

IV. ANALYSIS

GENERAL

The analysis section presented herein is in seven parts: Propulsion, Flight, Mechanics, Vehicles, Structures, Weights, Reliability, and Economics and Operations. Because of the number of study configurations, a Configuration Code was established to expedite the study. The Configuration Code is presented in Table 1.

Table 1. Study Booster System Codes

CONFIGURATION CODE

First Stage					Second Stage			
1	2	3	4	5	6	7	8	9
X	[N	(0.15)	J'	H] /	N	(0.2)	J	H

1	Number of Vehicles (when applicable)
2 and 6	Number of Engines (for two or more)
3 and 7	Engine Thrust (1 million x 10^{-6} pounds)
4 and 8	Engine Type (see Engine Code)
5 and 9	Propellant (see Propellant Code)

ENGINE CODE

Code Letter	Manuf.	ϵ_x	Designation	Type
B	NAA		Mod. J-3	Conv. Bell
E	NAA		Mod. E-1	Conv. Bell
F	NAA		Mod. F-1	Conv. Bell
J	NAA	27	Mod. J-2	Conv. Bell
J'	NAA	18	Mod. J-2	Conv. Bell
J''	NAA	13	Mod. J-2	Conv. Bell
P	GE			Adv. Plug
W	P and W	23		Adv. Hi-Press
W'	P and W	90		Adv. Hi-Press
W''	P and W	400		Adv. Hi-Press
Y				Fluorine

PROPELLANT CODE

Code Letter	Propellants
R	$LO_2/RP-1$
H	LO_2/LH_2
F	LF_2/LH_2
S	Solid

PROPULSION SYSTEMS

ENGINES

ENGINE TYPES SELECTED

The engines selected for study represent a cross section of the most promising designs and concepts for vehicles of the thrust range specified in the contract (600K to 3000K). Propellants considered were LO_2/RP-1, LO_2/LH_2, LF_2/LH_2, and a representative solid such as polyurethane or polybutadiene acrylic acid. The engines selected, which are under active development, are the F-1 and J-2. Engines not under active development at the desired thrust level were 150K (SL) LO_2/LH_2 engine (J'-2), 800K LO_2/LH_2 engine (J-3), a modified E-1 (600K), an LO_2/LH_2 plug nozzle engine, a segmented solid engine, the high pressure engine, and a LF_2/LH_2 engine. It was felt that some combination of engines and propellants in this group would represent a cost optimized boost vehicle from the propulsion system standpoint. Some of the advanced engine concepts, such as the reverse-flow nozzle and the rotating chamber engine, may show promise; however, sufficient data was not available on these systems to warrant consideration in this study. Nuclear propulsion systems also were not a subject of this study.

F-1 Engine

The F-1 rocket engine is a single-start, fixed-thrust, gimbaled, bipropellant system which uses liquid oxygen and RP-1. Basic components of the F-1 engine are a tubular-wall thrust chamber, a direct drive turbopump, a gas generator, and the controls associated with each. Fuel is used to cool the thrust chamber regeneratively, and as the turbopump lubricant and control system fluid. The high-pressure fuel is also used as the hydraulic actuating medium for gimbal actuators and various hydraulically operated valves in the engine system. The engine is turbopump fed and the pump is driven by a gas generator which uses the same propellant combination as the thrust chamber. The turbopump is mounted directly on the thrust chamber, thus eliminating the need for high-pressure flexible ducting. See Table 2 for performance details.

J-2 Engine

The J-2 rocket engine is a multiple-start, high-energy, upper-stage propulsion system which uses liquid oxygen and liquid hydrogen as propellants.

Table 2. F-1 Engine Design Parameters

Item	Description
Oxidizer	Liquid oxygen
Fuel	RP-1
Thrust (SL), (lb)	1,500,000
Thrust (vac), (lb)	1,725,000
Specific impulse (SL), (sec)	264.9
Specific impulse (vac), (sec)	304.9
Total propellant flow rate (lb/sec)	5,661.4
Mixture ratio (O/F)	2.25
Expansion area ratio	16
Chamber pressure (psia)	1,000
Characteristic velocity (ft/sec)	5,580
Length (in.)	222
Exit diameter (in.)	148
Weight (burnout), (lb)	15,000*
*Preliminary	

It has a single, regeneratively-cooled, tubular-walled thrust chamber. Independently driven direct-drive turbopumps, powered in series by a single gas generator, are used for the propellant feed system. The gas generator utilizes the same propellants as the main thrust chamber. Engine contact with fluids is limited to the propellants and helium. The latter is used as the actuating fluid for system controls. Gimbal actuators are not provided, but an accessory drive pad is furnished as a source of power for a hydraulic pump or other equipment. Flexible inlet bellows to the turbopumps are provided for attachment to rigid vehicle plumbing. See Table 3 for performance details.

J'-2 Engine

The J'-2 designation was assigned to a modified J-2 engine. The purpose of this investigation was to obtain an LO_2/LH_2 first-stage engine without incurring the cost penalty associated with developing a new engine. This was accomplished by modifying the exhaust nozzle on the 200K (vacuum) J-2 engine to obtain a sea level thrust of 150K. The combustion chamber and propellant pumping system were assumed to remain unchanged. According to Rocketdyne, the J-2 gas generator power cycle is flexible and a variety of nozzle area ratios can be accomplished with only a minor effect on engine performance or qualification. The J'-2 engine was assumed to have an

Table 3. J-2 Engine Design Parameters

Item	Description
Oxidizer	Liquid oxygen
Fuel	Liquid hydrogen
Thrust (vac), (lb)	200,000
Specific impulse (vac), (sec)	426
Total propellant flow rate (lb/sec)	470.21
Mixture ratio (O/F)	5.00
Expansion area ratio	27.50
Chamber pressure (psia)	632
Characteristic velocity (ft/sec)	7,379
Length (in.)	116
Exit diameter (in.)	80
Weight (burnout), (lb)	2,132

expansion ratio of 18 which provided a sea-level thrust of approximately 150K and a specific impulse of 320 seconds. A cluster of four engines was used for a vehicle thrust of 600K. A nominal development cost was added to account for the R & D required for the nozzle change.

J-3 Engine

The J-3 was established as an 800K sea level thrust LO_2/LH_2 pump-fed engine incorporating a conventional 80 percent bell nozzle and a bipropellant gas generator system. Engine design was assumed to be similar to the J-2 except that the combustion chamber pressure and nozzle area ratio were varied. The engine had a single, regeneratively-cooled thrust chamber utilizing a one-and-one-half pass cooling system with fuel as the coolant. Independently driven, direct-drive J-2 type turbopumps were used, and the gas generator was assumed to be capable of a tank head start. Pertinent performance details are listed in Table 4.

E-1 Engine

An LO_2/RP-1 engine in the lower thrust range (600K) was investigated to determine the effect of the lower energy—lower cost propellant combination on boosted payload cost. Since such an engine does not exist, the E-1 engine concept was used for the comparison. The E-1 was originally intended to develop 415K with growth possibilities to 500K. The engine is pump-fed, uses a regeneratively-cooled thrust chamber and has a bell nozzle. Present E-1 design details were not modified for this study due

Table 4. J-3 Engine Design Parameters

Item	Description
Oxidizer	Liquid oxygen
Fuel	Liquid hydrogen
Thrust (SL), (lb)	800,000
Specific impulse (SL), (sec)	344
Total propellant flow rate (lb/sec)	2,350
Mixture ratio (O/F)	5.0
Expansion area ratio	16
Chamber pressure (psia)	835
Length (in.)	181
Exit diameter (in.)	118
Weight (burnout), (lb)	11,000

to the minute effect of these parameters on payload cost. The performance data used was I_{sp} (SL) = 252 seconds, thrust = 600K, chamber pressure = 821 psia, expansion ratio = 12:1, and dry weight = 4,300 pounds.

Plug Nozzle

Four vehicles were investigated using the plug-nozzle engine. One was at a thrust level of 600K and the other at 1500K with both vehicles using LO_2/LH_2 and LO_2/RP-1 propellants. A combustion chamber pressure of 600 psia (slightly lower than that used with a bell-nozzle unit) was selected for this engine because the plug-nozzle engine requires high-pressure plumbing around the circumference of the engine to distribute the propellants to the annular injector. This results in an undue weight penalty when pressures become too high. On the other hand, the bell-nozzle engine requires a relatively small amount of high-pressure hardware.

An expansion ratio was selected which would result in an engine diameter that matched the tank diameter. This is possible due to the plug nozzle operating at near optimum expansion until the design area ratio is reached. Consequently, a relatively high expansion ratio was possible because the usual problems of over-expansion were not present. Selection of plug-nozzle area ratios are limited, however, by vehicle diameter, minimum injector width and internal expansion requirements. The latter parameter may be a limitation when differential throttling of the quadrants is used for thrust vector control.

The engines studied were assumed to be pump-fed and have a bipropellant gas generator utilizing the main propellants. A portion of the engine was

regeneratively cooled, and the plug itself was coated with an ablative material. Redundant pilot valves were used for controlling the main propellant and gas generator valve actuators. This was necessary to enhance the reliability level because a very large number of components are required with a plug-nozzle system. Thrust vector control was obtained by differential throttling of combustor segments. Table 5 lists the pertinent performance details.

Table 5. Plug-Nozzle Engine Design Parameters

Item	Description	
Oxidizer	Liquid oxygen	
Fuel	Liquid hydrogen	
Thrust (SL), (lb)	600K	1500K
Specific impulse (SL), (sec)	345	345
Total propellant flow rate (lb/sec)	1738	4350
Mixture ratio (O/F)	5:1	5:1
Expansion area ratio	24	24
Chamber pressure (psia)	600	600
Length (in.)	95	135
Diameter (in.)	163	240
Weight (wet), (lb)	6613	14,730

Segmented Solid

A single configuration was investigated using a solid motor for first-stage propulsion. The method selected for comparison was to hold payload constant and then optimize the first-stage configuration. The payload selected was the one obtained with the 600K liquid system. This resulted in a first-stage thrust level of 1400K. The purpose of comparison in this report is to present the relative boosted payload cost for vehicles with liquid and solid first stages.

Data supplied by members of the solid propellant motor industry were used to establish a vehicle configuration. The concensus of opinion in the solid motor industry is that the development of solid propellant powerplants in the 1400K thrust range may be accomplished by using current motor manufacturing methods, propellants, and materials. None of the data available fits the configuration established precisely; however, sufficient design information was obtained to perform the desired comparison. Basically, the data used were obtained from reports furnished by Aerojet, Thiokol, and Grand Central and from information supplied by Atlantic Research, Rocketdyne,

and United Technology Corporation. Many variations in parameters were encountered in surveying the data. For instance, where high-strength steel cases were used by one manufacturer, another used a filament wound case, and still another used a low-strength steel case. All claimed their selection represented the lowest cost configuration. Study results show, however, that due to the size of the vehicles and magnitude of the programs involved, slight variations in motor design did not significantly affect the payload cost. An upper burning time limit of 80 seconds was imposed on the solid-stage optimization. This figure is considered reasonable for the present nozzle state of the art. It is known that an increase in burn time would lower the thrust level, although no attempt was made to determine the optimum figure. Previous studies have resulted in an optimum T/W of 2 for large solid boosters, and this value was used in the study.

Motor mass fraction predictions varied from 0.89 to 0.82 depending upon whether thin, high-strength or thick, low-strength steel cases were used. Filament wound plastic cases provided high mass fraction and reasonable cost; however, steel cases appear more compatible with early program schedules. Manufacturing facilities for low-strength, non-heat-treated steel cases are readily available for any diameter under consideration. The high-strength heat-treated steel and reinforced plastic cases would require an expansion of present facilities. A figure of 0.85 was considered representative of an attainable stage mass fraction for a 1.4 $\bar{M}$ pound thrust vehicle with the required control system. This figure is somewhat lower than some of the more ambitious figures obtained from the industry; however, past experience with design and manufacture of large vehicles has shown that high mass fraction, low cost, and high reliability are not necessarily synonymous. The latter two items assume very significant importance compared to other parameters when large boost vehicles are studies; consequently, it was assumed that reliability and cost would not be compromised to attain high structural efficiency.

Thrust Vector Control

Various methods of thrust vector control were considered. Among these methods were auxiliary jets, jet vanes, swivel nozzles, jetevators, and secondary injection. Auxiliary jets, while being quite feasible and versatile, complicate the system by requiring a separate propellant supply. The overall effect was to make manufacturing, handling and launching procedures more complex while reducing stage reliability. Jetevators would require excessive weight due to thermal and abrasion problems unless some practical means of cooling could be devised. Swivel nozzles, while being simple and efficient, require large hinge moments and a high temperature seal. Though this system is considered quite feasible, the jet vane and secondary injection systems are deemed more practical for this application. Jet vane system reliabilities have been proven on various missiles. The system is simple and is current

state of the art. Although the weight is relatively high and a loss of performance is incurred, development and production costs are relatively low. The secondary injection system appears to be light, economical to produce, and efficient; however, application of this system to large boost vehicles would most certainly involve an extensive development program. A reasonable compromise seems to be use of the jet vane system to meet early vehicle availability requirements with a phasing in of the secondary injection system once its characteristics have been completely documented. Another method of vehicle attitude control which may be utilized for very large vehicles is the rotating grain solid propellant motor. NASA is currently sponsoring development of such a concept.

HIGH PRESSURE ENGINE

High combustion chamber pressure ($P_c > 1000$ psia) provides several advantages for a pump-fed liquid rocket engine. Among these are good low altitude performance, the capability of increasing high-altitude performance, small physical size, and possible weight reduction over the more conventional engines. Although these apparent advantages have been known for quite some time, previous methods used in attaining the high pressure have resulted in a substantial loss of performance due to the gas generator flow. A recent innovation in high-pressure propellant pumping systems, presented by the Pratt & Whitney Aircraft Company, appears to have reduced these losses to the extent that a high-pressure rocket engine may be highly desirable. It is this concept which was considered during the study. (A discussion of this concept may be found in Pratt & Whitney report FR-236.)

A high chamber pressure coupled with LO_2/LH_2 propellants results in a rocket engine with desirable characteristics for both first-stage and upper-stage vehicles. At sea level, the high pressure allows use of a higher expansion ratio to attain optimum expansion and a correspondingly higher specific impulse. At altitude, a gain in specific impulse is realized with the higher expansion ratio which is possible due to the small throat area, assuming nozzle exit diameter is limited by vehicle diameter. Although only the gains have been mentioned thus far, several problems accompany the use of high chamber pressure. Among these are pump development, combustion chamber and nozzle cooling, and engine gimbaling. The major problem associated with high-pressure rocket engine pumping systems is how to attain the high pressure without a significant loss in impulse. The Pratt & Whitney method involves the use of a preburner which burns oxygen and hydrogen at a very rich mixture ratio to provide power to drive the pumps. The preburner exhaust is then fed into the combustion chamber and completely burned with the addition of oxygen. Whereas conventional gas generator systems impose approximately a 1 percent loss in specific impulse per 1000 psia of chamber pressure, the losses incurred with a preburner system are considerably less.

The increase in combustion chamber temperature, due to the increase in chamber pressure, does not appear to present a serious problem. It is predicted that conventional cooling methods, such as regenerative fuel flow and film cooling, will be satisfactory. Attitude control of a vehicle utilizing the subject engine may present more of a problem and is discussed in more detail later in the report. The preliminary nature of the high-pressure engine concept allowed many variations of the engine to be studied. Single chamber thrust levels varied from 100K to 800K, and chamber pressure was varied between 1000 and 5000 psia. Although this does not represent the upper limit for the high-pressure concept, detailed parametric data was available for this thrust range. Single and multiple chamber installations were investigated as were primary and secondary expansion systems.

Liquid Fluorine/Liquid Hydrogen Engine

Parametric data was not readily available for fluorine/hydrogen engines at thrust levels from 100K to 300K. Consequently, the data developed for smaller engines was used to establish design and performance figures for the LF_2/LH_2 upper stage engines used in this study. The engines were assumed to be pump-fed, have multipass, hydrogen cooled thrust chambers, and utilize the expanded hydrogen method of powering the pumps.

A delivered specific impulse of 456 seconds was used assuming a combustion efficiency of 97 percent and a nozzle efficiency 98.3 percent. An expansion ratio of 45 was used to attain this figure of specific impulse. A combustion chamber pressure of 600 psia was selected to provide a reasonable engine size yet not complicate pump requirements. Engine dimensions and weights were determined from empirical formulas developed by S&ID. (See Table 6.)

Table 6. LF_2/LH_2 Engine Design Parameters

Oxidizer	Liquid fluorine
Fuel	Liquid hydrogen
Thrust (vac), pound	300K
Specific impulse (vac), second	456
Total propellant flow rate, pound/second	658
Mixture ratio, oxidizer/fuel	13:1
Expansion area ratio	45
Chamber pressure, psia	600
Length, inches	178
Diameter, inches	128
Weight (wet), pound	3924

Bell Nozzle

Engines considered during the study which utilize the conventional bell nozzle are the F-1, J-2, E-1, and the LF_2/LH_2 engine. This type engine has been under development since the inception of the rocket engine industry, a fact which indicates that certain advantages are accrued in utilizing such a configuration. Some of these are ease of construction, installation and thrust vector control, relatively good performance at low and high altitudes, and relatively low weight due to the small amount of high-pressure hardware necessary. Various advanced nozzle concepts offer advantages in one or more areas of design or performance, but the bell-nozzle engine appears to provide the best compromise overall. The high-pressure engine is essentially a bell-nozzle engine but was considered separately due to its unique pumping cycle.

Conventional bell-nozzle pump-fed engine chamber pressures usually run from 600 to 1000 psia, and the pumps generally are powered with a bipropellant gas generator utilizing the main engine propellants. The high-speed pumps provide a convenient auxiliary power source and the engines may be gimbaled by hydraulic or electrical means.

Plug Nozzle

Pump-fed plug nozzle performance initially appears to be higher than that of an equivalent bell nozzle; however, detail investigation shows that certain penalties are incurred with the performance increase. At thrust levels from 600K to 1500K, pump-fed plug-nozzle configurations generally provide slightly lower sea-level performance than a bell-nozzle engine for a given thrust level and propellant combination. This is due to the trade-off between chamber pressure, engine weight, and performance. Specific impulse at low altitudes is dependent upon combustion chamber pressure among other parameters. Nominal bell-nozzle chamber pressures at high thrust levels run from 800 to 1000 psia while plug nozzle chamber pressures run from 600 to 800 psia. The reason for the difference in pressures is that the plug nozzle requires high-pressure plumbing around the circumference of the engine to distribute the propellants to the annular injector. This results in an undue weight penalty when pressures become too high. On the other hand, the bell-nozzle engine requires a relatively small amount of high-pressure hardware. Consequently, in order to be competitive from the weight standpoint, a lower pressure is generally used with a plug-nozzle engine and a loss in specific impulse is incurred. This loss is minimized due to the plug nozzle operating at near optimum expansion ratio up to the design pressure ratio and, as a result, the average specific impulse for a given trajectory is only slightly below that of a bell-nozzle engine.

Two advantages of the plug-nozzle engine were readily apparent. The first was that plug-nozzle reliability could reach a high level economically through component redundancy. Although a very large number of components are necessary for a large engine, some finite amount of component malfunction could be tolerated before a serious performance decrement would be incurred. The second advantage was concerned with the development of an engine substantially larger than the F-1. Should the need for such an engine arise, it appears that a strip injector-type engine would present less design and development difficulty than a circular injector-type engine. This conclusion is based on relatively limited analytical and test data for the strip injector, whereas the combustion instability problems of circular injectors are well documented.

ENGINE PERFORMANCE AND APPLICATION

High Pressure Engine

The relative advantages of high chamber pressure have been discussed previously. The values of pressure investigated during this study were 1000, 3000, and 5000 psia. One of the first steps during the study was to determine the most desirable combustion chamber pressure for the vehicles to be studied. Perusal of the estimated performance curves generated by Pratt & Whitney showed that, for optimum sea-level expansion, a specific impulse increase of 7.4 percent would be realized by increasing chamber pressure from 1000 to 3000 psia while only 2.8 percent increase would be obtained between 3000 and 5000 psia. The vacuum impulse is approximately the same for all three pressures providing a secondary expansion system is used and the area ratio is the same for all engines. If this is not the case, the higher-pressure engine will provide the higher-vacuum impulse due to optimum sea-level expansion occurring at a higher area ratio.

Engine weight increase for the same variation in chamber pressure is considerably more prominent. For instance, a 29.2 percent weight increase is realized for a chamber pressure increase between 1000 and 3000 psia, and a 57.2 percent increase accompanies a chamber pressure increase between 3000 and 5000 psia. The preceding comparison is presented for a thrust level of 600K and considers bare engine weight (plus boost inducer) only. It is recognized that the gimbal actuator system and thrust mount weights would also increase. The effect of this weight increase was to decrease the booster mass fraction.

Payload comparisons were then made for the 3000 and 5000 psia values. The 1000 psia system was not used because the nozzle exit diameter would exceed the vehicle diameter when the very high expansion ratios were used. The payload difference was only 2.6 percent with the 3000 psia engine providing the higher payload. This small difference does not make chamber pressure selection obvious. Estimated development cost of the 5000 psia engine is 16.2 percent higher than that of the 3000 psia engine while

production costs at a nominal production rate are 7.83 percent higher for an 800K engine. Development time is more a function of thrust level rather than chamber pressure so these development times were assumed to be identical.

This discussion indicates that an increase in chamber pressure (above 1000 psia) provides a higher average specific impulse, an increase in engine system weight and an increase in development and production costs. Since the 1000 psia value was eliminated due to the extremely large exit diameters at very high expansion ratios (i.e., ~ 40 feet at $\Sigma = 400$, single stage-to-orbit case), the selection was between 3000 and 5000 psia. The payload comparison showed the 3000 psia system provided the higher payload although the difference was slight. In addition, the 5000 psia system costs were generally higher. As a result, the 3000 psia system was selected for further configuration study.

This report shows that a high-pressure engine utilizing the preburner concept presents some installation and control problems. The usual method of controlling thrust vector is to gimbal the engine or nozzle. With this engine, the gimbal point may present a problem. Figure 4 shows that stacking of the components results in the least performance losses and the least weight. However, it does not appear that the rotating machinery housings would accept the loads imposed during gimbaling without excessive deflection, except in the area of the ejector. This area, however, is located near the midpoint of the engine and Figure A-1 shows one possible method of gimbaling at this point. One problem with this method is the deflection of the propellant lines. An unacceptable amount of deflection is encountered unless the flex joints are located near the gimbal point, which would result in increased system weight and increased line losses. Another approach to the problem is to relocate the components. Discussion with Pratt & Whitney indicates that this is entirely feasible and that Figure A-1 represented the best installation from the engine manufacturer's standpoint. Possible modifications include parallel or horizontal mounting of the preburner and pumps, both of which provide a reduction in engine length. The preliminary nature of the engine concept did not allow detailed analysis or design of a gimbal system. It is felt, however, that gimbaling of the engine in most any configuration is possible.

Various other methods of vehicle attitude control are available, however. Jet vanes, secondary fluid injection, a small attitude engine system, and rotatable solid motors are a few of these methods. One of the more interesting systems for large boost vehicles is the rotatable solid. With this system several solid motors are utilized with their nozzles pointed aft to burn throughout the boost period. Orientation of the nozzles is such that pitch, yaw, and roll control is obtained when they are rotated, and when they are pointed aft they add to the overall vehicle impulse. Such

a concept is under active development at this time. It was assumed, for the purpose of this study, that the high pressure engine was readily adaptable to a gimbal system. The use of secondary expansion, as in the single-stage to-orbit case, precluded engine gimbaling, however. Differential throttling of the modules, or one of the previously mentioned methods, may be used with this configuration.

High-pressure engine performance estimates furnished by Pratt & Whitney used a nozzle and combustion efficiency factor of 95.6 percent. These figures were further reduced to accommodate vehicle requirements. Among these requirements are propellant bleed for tank pressurization, LOX pump-power take-off pad for hydraulic pump and generator requirements, and a power take-off to drive the gear driven boost inducer system for both the fuel and oxidizer. This approach is considered conservative since all these items may not be required for any one vehicle. A constant mixture ratio of 5:1 was used although it is recognized that the optimum mixture ratio varies with expansion ratio. One exception was during the investigation of the effect of mixture ratio variation on payload. This investigation showed that a change in O/F from 5:1 to 7:1 provided a decrease in payload, although the decrease was slight. This change was brought about by an approximately 2 percent decrease in specific impulse and an increase in propellant bulk density from 0.01165 pound per cubic inch to 0.0142 pound per cubic inch. Consequently, the 5:1 O/F ratio was used throughout the study due to the performance advantage and due to the lower combustion chamber temperature associated with the lower mixture ratio.

A first-stage engine module expansion ratio of 23 was used in most cases. The nozzle was assumed to be conical although a contoured nozzle would provide shorter engine length and possibly higher nozzle efficiency. Second-stage expansion ratios were varied between 45 and 90. Here, again, the conservative approach was used and the numbers selected were arbitrary. The high-pressure engine presents the possibility of utilizing expansion ratios limited only by vehicle diameter. The area ratio value of 45 was investigated for use in both stages as a means of reducing engine production costs. All three engines (2 first stage, 1 second stage) could be identical. The figure of 45 is purely analytical and is based upon nozzle separation criteria established by S&ID. It was assumed that nozzle separation would not be allowed.

One of the more interesting configurations investigated was the single-stage-to-orbit case. The performance of this vehicle was based on the assumption that an engine system utilizing secondary expansion could be developed. This system consists of several high-pressure engine modules clustered and incorporating multiple pumps or a single pump for all modules. The expansion ratio of the modules would be 23. A secondary expansion

surface is provided in the form of a large single bell nozzle. The area ratio in this case was assumed to be 400. If ambient pressure can be maintained in the center of nozzle, optimum expansion will occur until the design area ratio is reached. Flow initially will be in annular shape at the nozzle exit. As the back pressure is reduced, the nozzle would then flow full. There may be some question as to the method to be used for maintaining ambient pressure in the center of the nozzle. Considerable testing would be required to fully document the performance of such a configuration.

The high-pressure engine cycle is estimated to require from 50 to 100 psia at the pump inlets. Such magnitudes of pressure cannot presently be supplied by pressurization systems without undue weight penalties. One solution to the problem is either tank-mounted or engine-mounted boost pumps. The latter method was assumed for this study since weight data for such a system was provided by Pratt & Whitney. The boost inducer system envisioned by Pratt & Whitney is gear-driven and powered by the preburner rotating machinery. The exact performance degradation associated with this system have not been determined as the pump NPSH requirements are not established. Theoretical performance quotes were reduced a nominal amount, however, to account for combustion efficiency, propellant bled for pressurization purposes, and power extracted to drive the boost inducer system.

Solid Propellant

Combustion chamber pressure selection is the one parameter the various manufacturers most nearly agreed upon. Values were quoted from 700 to 1000 psia. The most frequent figure was 900 psia. A decrease in chamber pressure did not provide the increase in propellant-to-motor weight ratio that might be expected due to the increase in nozzle weight. The nozzle weight for an 80 second motor is a significant contribution to the total motor inert weight.

Delivered specific impulse values from 235 to 245 seconds were quoted for the various motors depending upon the chamber pressure and type of propellant used. A reference specific impulse of 245 seconds and a delivered specific impulse of 240 seconds was used for the performance calculations in this study. Jet vanes, a chamber pressure of less than 1000 psia, and off-optimum expansion ratio accounted for the 5 second loss. The propellant formulations in most all cases were tried and proven and consisted of raw materials which were quite abundant. In general, high density propellants (greater than 0.07 pound per cubic inch) will give low performance and high cost. The high performance formulations (greater than 250 seconds) also result in high cost and were in no instance recommended by the motor manufacturers contacted.

Of the two methods of construction offered (segmented and unitized), the segmented motor was considered the most desirable for such large units. Although the reliability and mass fraction is slightly lower for a segmented motor, important advantages are accrued in terms of quality control, logistics, and manufacturing facilities. In addition, less loss is incurred when rejects are encountered. Statistics from one manufacturer showed that approximately 10 percent rejects had resulted from past motor manufacturing experience. It was assumed that propellant used in such large motors could be cured at temperatures from 80 to 110 F. Another advantage of the segmented motor is that the propellant manufacturing and casting facility does not need to be located in the proximity of the launch site, a fact which allows use of present facilities. This would be required for the unitized motor unless some complex method of water handling and transportation were used.

ENGINE DEVELOPMENT

Problems associated with engine development depend largely upon size and somewhat upon type. Development progress of the F-1 and J-2 is quite well known and will not be discussed in this report. One major problem associated with high-thrust engine development is the power source for pump testing. In the case of the high-pressure engine, concurrent testing of the pumps and preburner would be highly desirable to reduce development time and cost. A workhorse gas turbine of some sort may solve this problem with the preburner being phased in for integrated testing at some later date. If a configuration such as Figure A-2 were used, considerable scale model testing would be required to fully document the altitude performance of the large secondary expansion nozzle. This has been proven practical in the past, providing scale-up factors can be determined. The engine development picture would be further complicated if differential throttling of the modules was used for thrust vector control of the configuration due to the additional injector and combustion chamber problems.

Development time is, of course, largely determined by thrust level. It is estimated that 37 months would be required for a 250K high pressure engine. The segmented solid motor would require approximately the same length of time at the 1400K thrust level. The 200K LF_2/LH_2 engine would require 30 months. These figures are preliminary estimates but could serve as a guide for program planning. Actual times would depend upon the type of funding and the work load on present facilities. The engines for the single-stage-to-orbit vehicle will require additional time and probably new facilities. Another problem associated with this configuration is phasing. The engine modules would have to be in an advanced state of development before meaningful tests could be run on the secondary expansion nozzle. This could easily add 18 months to the development period. Development

periods of from 30 to 36 months were quoted for the segmented solid motors under consideration and these figures appear to be realistic in view of the past performance of the industry.

Engine development costs are subject to considerable conjecture and discussion. The estimates received from industry were increased slightly to reflect the vehicle contractor's handling costs. These costs are explained in detail in the Economics Section. It was not possible to obtain specific costs for all of the engine configurations considered. Consequently, most of the data was received in parametric form. This required S&ID estimates to cover desired variations in some of the engines. By the same token, it is recognized that the quoted costs would no doubt change if firm costs for a specific engine were desired. Past experience has shown engine development price estimates to be optimistic. A relative comparison, however, shows that for a vehicle utilizing two 750K high-pressure engines for the first stage and one 750K high-pressure engines for the second stage, a 50 percent increase in engine development costs results in a 3 percent increase in payload cost for the 100-launch vehicle program. This comparison could be made for most any major cost category and the results would be similar. The fallacy is that in reality, engine development precedes the conception of complete programs by a substantial time interval, and as a result, engine funding is frequently not considered as part of the overall program cost (F-1, J-2). One reason for this is that a given engine may be used for several programs although this has not generally been the case in the past.

[redacted]

PROPELLANT SYSTEMS

PROPELLANT PROPERTIES

Evaluation of boosters were based on the following propellant combinations:

Liquid Oxygen-Liquid Hydrogen (LO_2/LH_2)
Liquid Oxygen-RP-1 (LO_2/RP-1)
Liquid Fluorine - Liquid Hydrogen (LF_2/LH_2)

No particular problems are anticipated with any of these combinations in respect to material compatibility. Liquid fluorine does not appear to present compatibility problems as one might expect. Stainless steel 18-8 series, monel, copper and aluminum 17, 24, and 52 series have been successfully used with liquid and gaseous fluorines. The corrosion resistance of all material of construction used with fluorine depends on the formation of a passive fluorine film. Therefore, the only additional requirement necessary to prepare hardware for service, which ordinarily would not apply to other cryogenic propellant hardware, is passivating of the material. Ground handling of liquid fluorine would present a problem due to fluorine's extreme toxicity. Disposal of fluorine vapor from tank boil-off would undoubtedly include such items as fluorine-hydrocarbon-air burner, scrubber, and stack to prevent undue exit hazards.

Table 7 gives the properties of the propellants which were used in this study.

FEED SYSTEMS

Design and installation of line and components necessary to handle the propellants should present no unusual problems with boosters within the scope of the study. Propellant velocities up to 46 feet per second have been used as the governing criteria for establishing line sizes for LO_2, LF_2, and RP-1. Hydrogen line sizes have been based on velocities of 60 feet per second. Although these velocities may not be commensurate with minimum line losses, they represent a compromise of system weight with respect to residue fuels, structural requirements, and pressurant requirements, and compromise of system design in such areas as flexible lines.

Manifolding of propellant tanks is an important consideration when comparing lateral staging or drop tank concept with tandem staging. Intertank feed systems between jettisonable laterally staged tanks is necessary to assure

Table 7. Propellant Properties

Property	LF_2	LO_2	LH_2	RP-1
Molecular Weight	38	32	2.016	172
Melting Point F	-365	-362	-434	-50 to 100
Boiling Point F	-307	-297	-423	422
Heat of Vaporization Btu/lb	74	92	195	125
Vapor Pressure psia	14	15	16	0.05
Critical Temperature F	-201	-182	-400	758
Critical Pressure psia	808	731	188	315
Density Lb/Cu. ft.	94	71.3	4.4	50.4
Viscosity Centipiose	0.25	0.18	0.013	0.104
Specific Heat Btu/lb-F	0.367	0.406	2.45	0.50
At boiling point of liquid				

optimum propellant utilization. The drop tank concept not only requires intertank manifolding but an additional requirement that manifolding lines be capable of being disconnected at propellant depletion. System complexities because of propellant, pressurization, and electrical disconnects results in an overall reliability degradation. Single-tank tandem staging requires no additional manifolding since the tank acts as a manifold. Propellant loading systems associated with large boosters require auxiliary vent valves for LO_2, LF_2, and LH_2 during fill operations to allow for high initial propellant boil-off created by cool-down of propellant tanks during fill. In the case of multiple-tank modules, intertank propellant manifolding is desirable to allow all tanks to be filled simultaneously, thereby eliminating intertank temperature differential problems.

Recoverable vehicles require additional venting of residue propellants prior to the landing phase. The depletion of residue propellants should be accomplished at the end of boost and during the deacceleration phase. Fuel tank propellant residue would be drained or vented from an opening in the forward end of the tank. The LO_2 tank would be vented at the aft end and depletion of residue LO_2 complished by allowing the liquid to vaporize.

PROPELLANT UTILIZATION

A problem in the design of boosters is the tendency for one of the bipropellants to reach depletion before the other as a result of random variations in the engine and vehicle systems. In some cases significant quantities of otherwise usable propellants may be left in the vehicle causing lower velocity at burn-out. Propellant utilization normally refers to a system that

achieves simultaneous depletion of both propellants. The open- and closed-loop method may be employed to control the propellant utilization system.

These methods are discussed in the 1-20 M LB Boster study AF 04(611)-6069 and no further discussion is deemed necessary.

For large LO_2/RP-1 boosters, the closed-loop propellant utilization system is not considered necessary. The gain in performance associated with closed-loop propellant utilization for low-performance booster is not considered adequate to justify the degradation in overall reliability and cost. The accuracy of tank loading would then be the determining factor in overall mission success. Propellant tank accuracy has been developed to a point where loading tolerances of ± 0.25 percent of the tanks mass can be reasonably expected. Engine performance repeatability has improved due to improved engine calibration techniques. Fuel biasing will also tend to cut down the weight of propellant residue. In consideration of propellant utilization accuracy for first-stage RP-1 boosters, the closed loop system would not be recommended.

Comprehensive studies conducted in support of the Saturn S-II program, indicated a benefit of a closed-loop propellant utilization system for upper stages in a corresponding increase of payload capacity for a given probability level. This benefit is due to minimization of propellant residue over open-loop systems. Another method of minimization of propellant residue is accomplished if tank propellants are biased to increase the lesser propellant mass by a small amount. For a LO_2/LH_2 with a nominal mixture ratio of 5.0 the effects of fuel biasing can be considerable. As an example, if propellant loading is based on nominal in-flight mixture ratio, with considerations for known variables, equal probability of LO_2 or LH_2 residuals results. With a mixture ratio of 5 the average LO_2 residue would be 5 times as heavy as the LH_2 residue. Biasing the propellant to include more LH_2 will increase the amount and frequency of fuel residues, but will decrease the amount and frequency of LO_2 residue. Since 1 pound of either type of residue has the same effect on mission capability, the result is to minimize the average residue or summed residue of a particular number of flights.

An illustrative example of the effects of various propellant utilization methods are shown in Figure 5. The plot is the result of statistical and probability studies conducted for the Saturn program. Although the velocities shown relate to the Saturn S-II vehicle, the comparative benefits of closed-loop versus open-loop propellant utilization systems as shown would pertain to this study. Figure 6 shows an optimization for maximum payload transport per given number of launches (average pounds per launch) in which lines of constant payload in orbit have been plotted on coordinates of payload size and payload transport reliability. Transport reliability is the reliability of successfully completing the boost phase and does not include reliability of

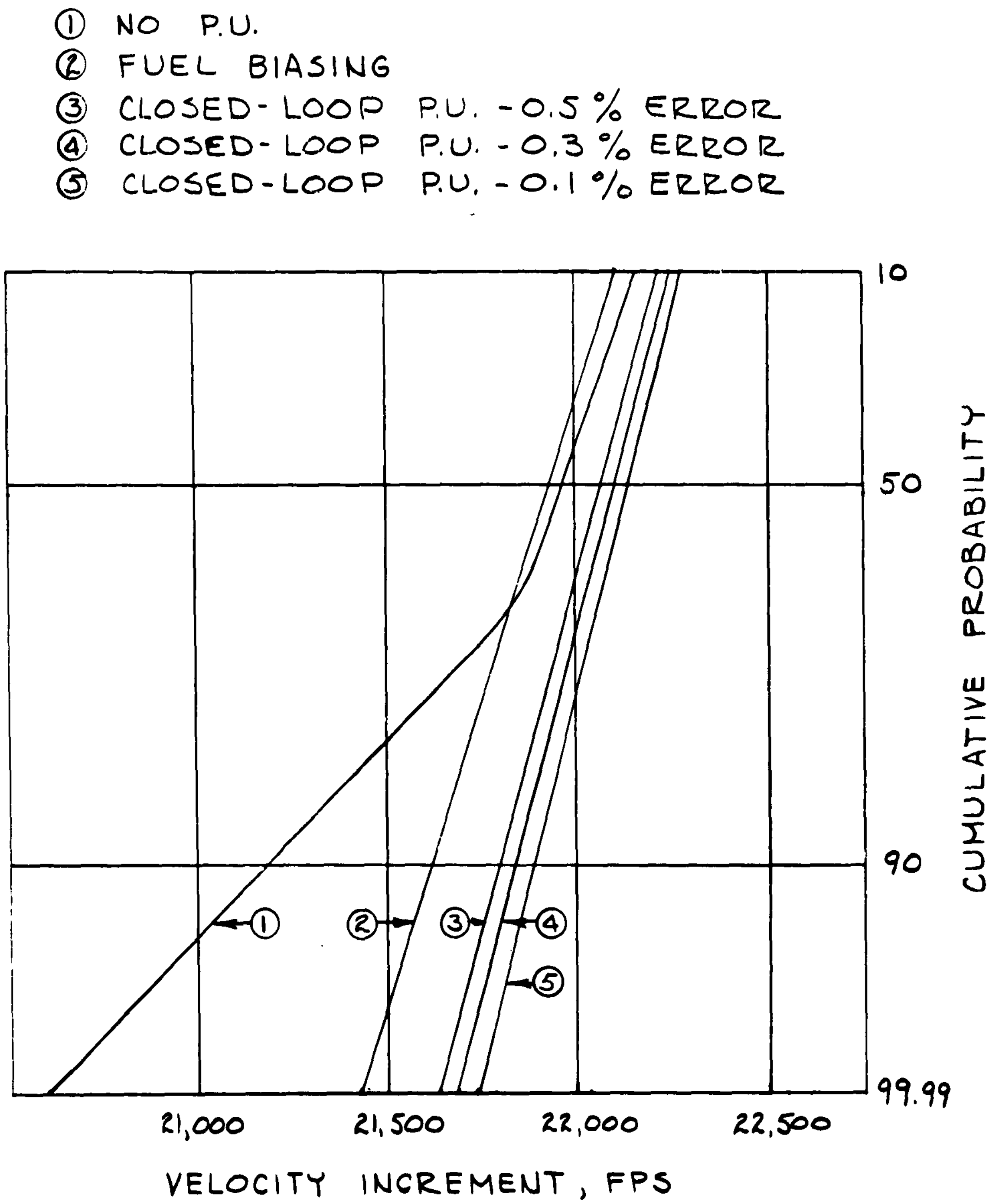


Figure 5. Probability of Achieving Various Velocity Increments (S-II Vehicle)

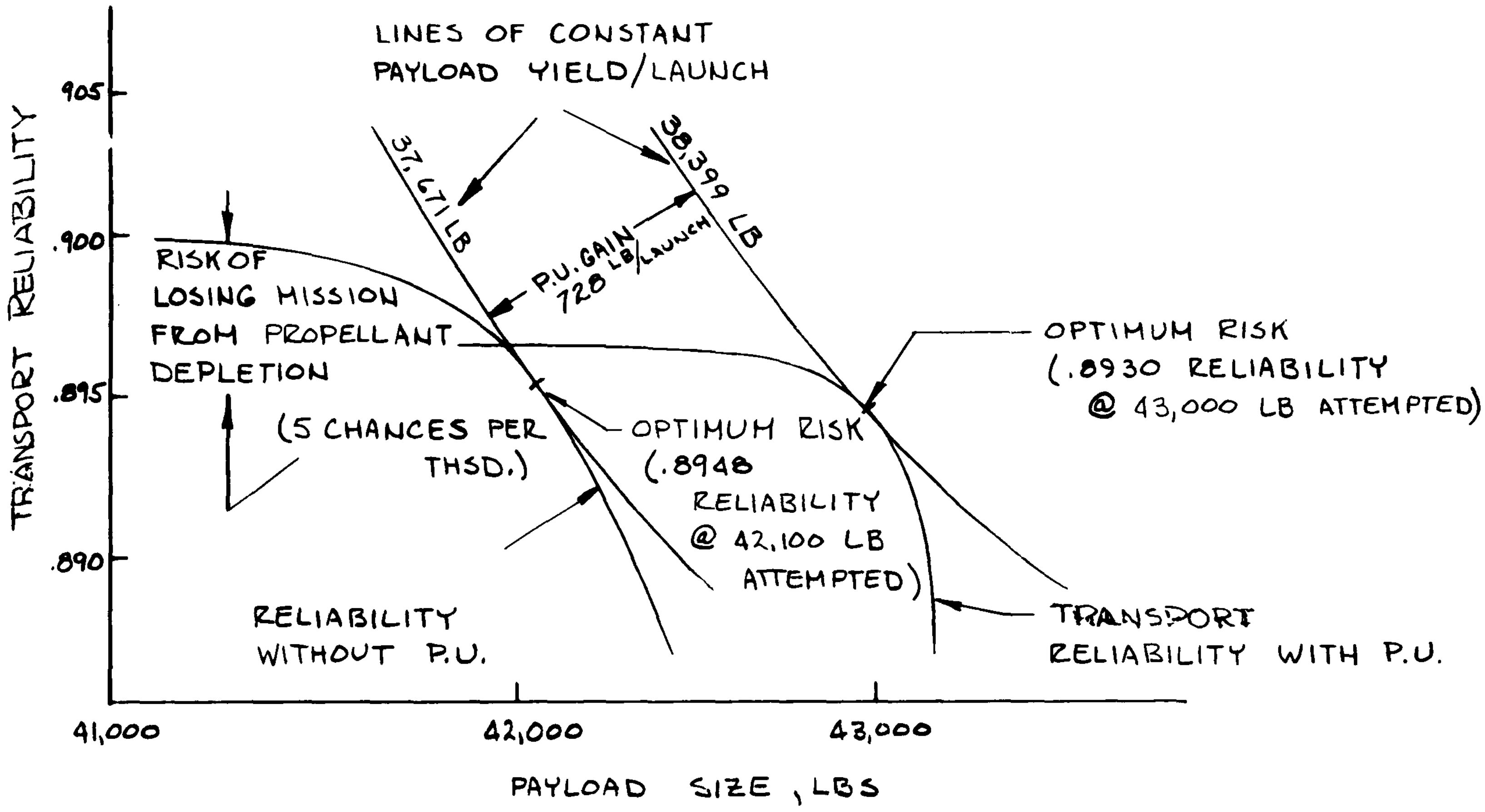


Figure 6. Effect Of Propellant Utilization On Payload Yield Per Launch

putting the payload in orbit. Also plotted is payload yield versus launch reliability at various payloads, with and without propellant utilization. The curves have been based on required S-II burn-out velocity with 0.900 nominal payload transport reliability and includes the variations of first-stage boost. A 0.997 functional reliability for a closed-loop system has been established for S-II studies.

Possibly one of the chief values of a closed-loop system is in accommodating unforeseen payload size requirements without prohibitive loss in mission reliability. Fuel biasing, plus other refinements in open-loop techniques appear to approach the advantage of closed-loop propellant utilization.

PRESSURIZATION SYSTEM

REQUIREMENTS

To satisfy turbopump operating requirements, it is necessary to provide some means of supplying the propellants under pressure to the pump inlets. Pump inlet pressure requirements are a function of the turbopump design and vary with the individual engines. Two common methods of supplying this pressure to the turbopump inlets are by boost pump or ullage pressure.

The use of boost pumps is attractive because they reduce tank pressurization, reduce line insulation needed to maintain NPSH requirements at end boost, reduce problems of loss of turbo NPSH from heat leak, and may permit use of a high-pressure turbopump of higher speed and less weight.

It is recognized that the use of boost pumps does not delete the pressurization system since it is desirable to maintain propellant vapor pressure; however, the weight of the system is decreased and in some cases the weight decrease can be considerable. Tank structural weights will be reduced, but the decrease in weight is not a direct function of ullage pressure since the tank skin thickness is also a function of longitudinal loads, as well as hoop tension. In no case can the ullage pressure be lower than ambient due to the weight penalty suffered by the propellant tank compressive loads. Structural weight savings are dependent on tank shape. In the case where the tank is approaching a spherical shape weight, savings can be considerable. When cylindrical tanks have a high fineness ratio, savings may only be a fraction of the spherical tank of equivalent volume. Disadvantages of a boost pump system is the added complexity. Reliability data is nonexistent for boost systems capable of handling the flow requirements for the thrust range of engines considered in this study. In the case of future engine development, such as the high-pressure engine, the design of the engine pump inlet net positive requirement should be taken into consideration during propulsion subsystem design optimization. Figures 7 and 8 illustrate examples of savings in weight by the application of boost pumps with low NPSH.

For the purposes of this study, any additional pressures needed to satisfy turbopump NPSH requirements, above that which is supplied by the minimum liquid acceleration head, is supplemented by ullage pressure. Table 8 is the NPSH requirements for the individual engines that were established as the controlling criteria, which determine the pressurization requirements.

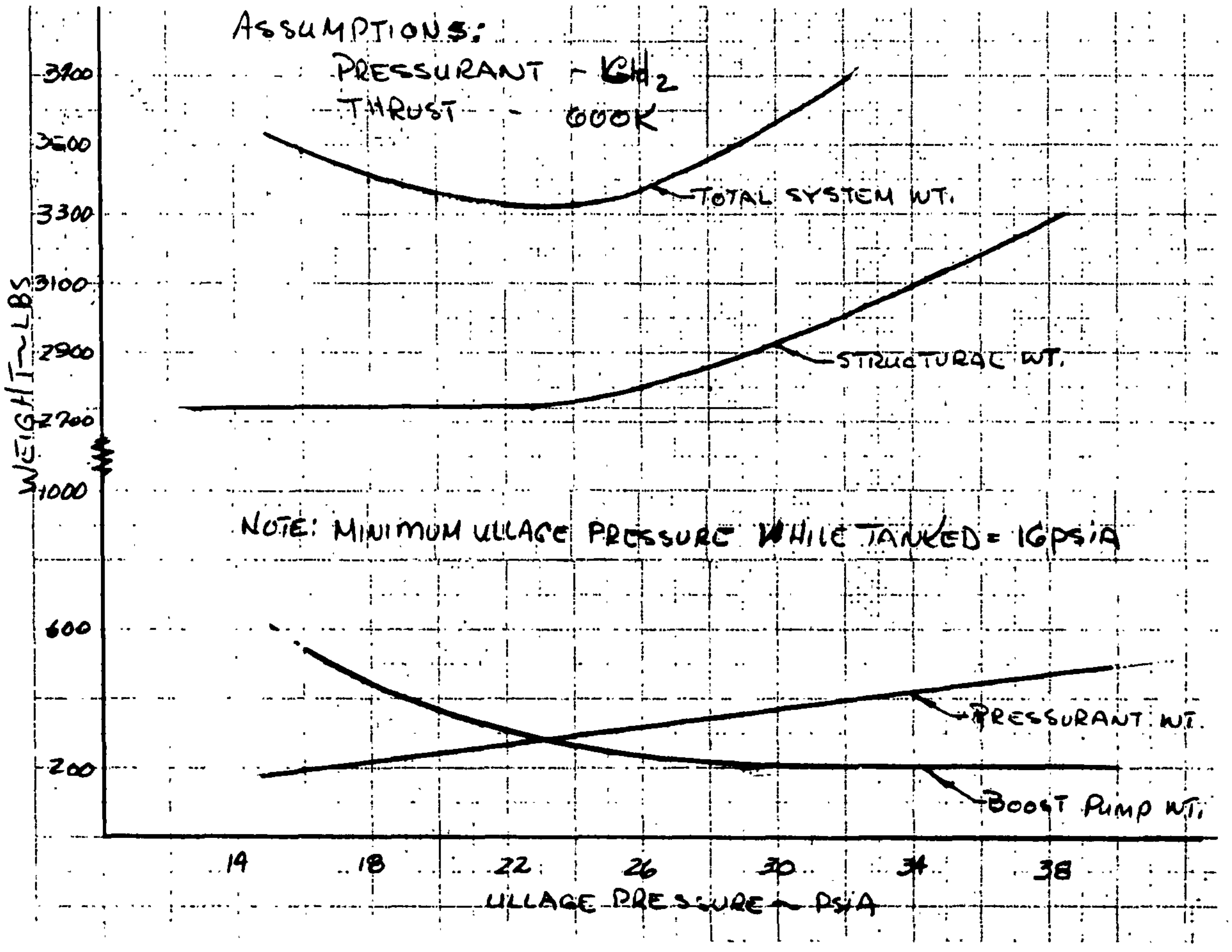


Figure 7. Boost Pump Effect On LH_2 Tank and Pressurant Weight

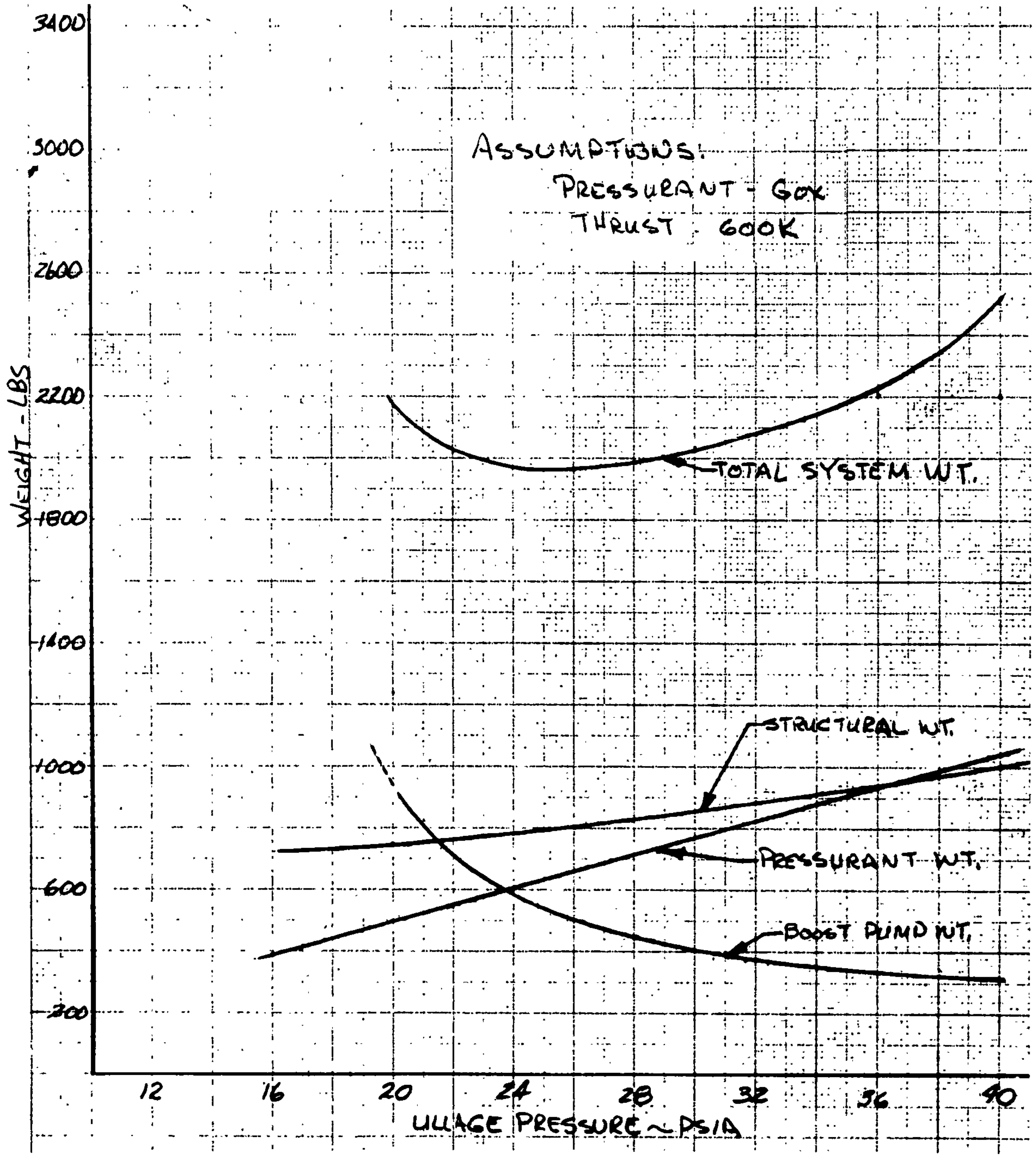


Figure 8. Boost Pump Effect On Gaseous Oxygen Tank and Pressurant Weight

Table 8. Net Positive Suction Head Requirements

Engine	NPSH FEET	
	Oxidizer	Fuel
F-1	71	110
1500K (LO_2/LH_2)	71	300
J-2	25	130
P&W (High-pressure engine)	25	130
Plug Nozzle	30	110
300K (LF_2/LH_2)	25	130

The design of pressurization systems in the past decade has generally been controlled by availability of space, heat source, and, most of all, engine operating parameters. With the possibility of development of a new engine, consideration should be given to the type and method of pressurization that will be associated with it. Each engine, regardless of size or intended use, should be capable of supporting its pressurization requirements. In the event engines are clustered to provide the necessary thrust, each engine would provide a portion of the necessary pressurization requirements.

PROPELLANT PRESSURIZATION SYSTEMS

Several methods have been used for pressurizing propellant tank ullage pressures. In the course of the study five basic system concepts were evaluated. These were: solid grain pressurization, high-pressure gas storage, low-pressure storage (liquid pressurant) with vaporization, products of combustion, and propellant vaporization. In comparing these systems, emphasis is directed toward the high-pressure engine with the thought that more flexibility can be assumed in regard to the engine's role in support of the pressurizing system.

Solid Grain

Recent development of the self-generating gas pressurization system, which generates gas below 1000F, has been accomplished by the Aerojet-General Corporation. System weights are comparable to liquid nitrogen evaporative system. At present, the azide formulations used for the generation of N_2 gas are expensive and not commercially available. Since obtaining nitrogen from a liquid nitrogen evaporative system is much more economical than from the azide grain, no further consideration was made for the solid grain system.

High-Pressure Gas Storage

High-pressure gas storage has a definite advantage in ground handling characteristics over most other pressurization systems which require separate pressurant storage. Basically, the system is relatively simple and consists of storage bottle, regulator, heat exchanger, and control valve. Storage of gas at ambient temperatures does create problems with heat exchanger design since adiabatic expansion of the gases results in progressively lower gas temperatures at the entrance of the heat exchanger, and a complex heat exchanger unit would be required to handle the variable mass flow rate of pressurant. When a pressurant with low boiling temperature is used, storage of the gas at liquid hydrogen temperatures will result in considerable weight savings. Also, variations in heat exchanger gas inlet temperature will be small since operation at a low temperature (200F) results in the process being essentially isothermal.

Low-Pressure Storage

Since minimum storage capacity with maximum available pressurant occur with the pressurant in its liquid state, liquid pressurant storage with pressurant vaporization capabilities results in a comparatively light-weight system. Certain handling problems are associated with such a system, particularly when cryogenic pressurants are used. Means of expelling the liquid pressurant from the storage container is necessary. A separate gas supply or vaporization of the pressurant within the storage container will supply the required expulsion pressure. Figure 9 shows a low-pressure storage system in which the heated pressurant gases are used for vaporizing the stored pressurant. In such a system as shown, the latent heat of vaporization should be low (such as helium) since the size of a heat exchanger for pressurant vaporization is limited. A comparison of stored liquid pressurant with other systems is shown in Figure 9.

Propellant Evaporative System

Vaporized rocket engine propellants for pressurizing tank ullage results in one of the most desirable pressurization systems. The reliability of a propellant evaporative system is high because of the elimination of a separate storage container and its accompanying controls. LO_2 and LH_2 propellants are readily adapted to an evaporative system. In a LO_2/LH_2 booster, the use of hydrogen on the liquid hydrogen presents the optimum system. The evaporation of LO_2 for oxidizer tank pressurization is attractive, and although not the lightest, it represents the optimum system for reasons which are discussed under Selected Systems in this section. RP-1 fuel does not lend itself to an evaporative system because of its relatively high latent heat of vaporization and high condensation temperatures. Liquid fluorine,

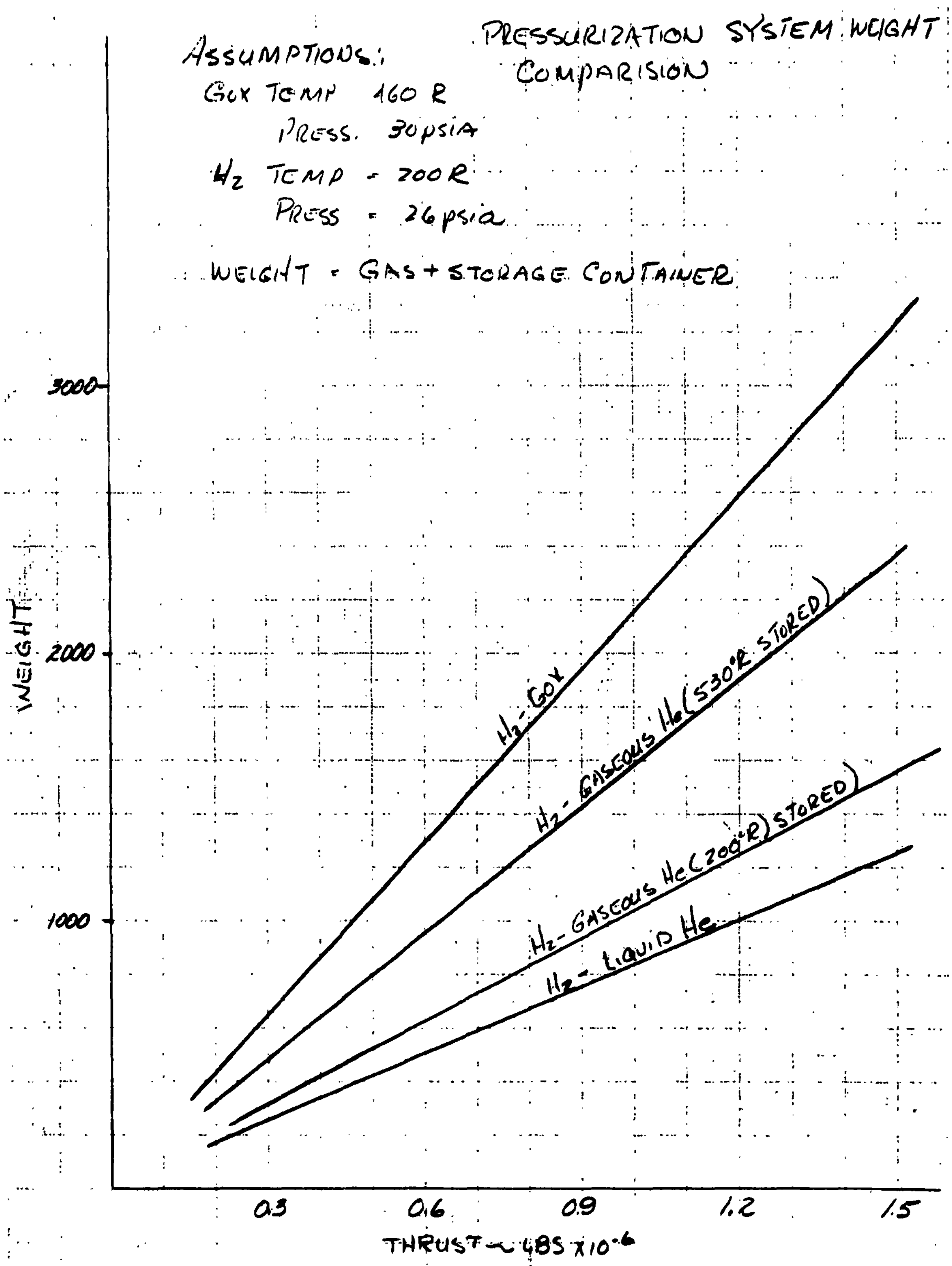


Figure 9. Pressurization System Weight Comparison

although adapted to an evaporative system, is not recommended because of its high molecular weight.

Products of Combustion

A promising method of propellant tank pressurization is to bleed the thrust chamber or gas generator products of combustion to propellant tank ullage. Such a system produces condensable gases or gases not completely inert to both fuel and oxidizer. In a case where RP-1 is used as a fuel, gas generator gases (normally fuel rich) bled to the RP-1 tank present a workable and desirable system. A heat exchanger is necessary to cool gas generator gases to avoid excess pressurant temperatures. In a LO_2/LH_2 engine system, a product of combustion system combined with an LO_2 evaporative system results in a comparatively simple system. The advantage of such a system is that one heat exchanger serves two purposes, the cooling of gas generator gases and the vaporization of LO_2. Products of combustion method of pressurization is limited to hydrocarbon fuels and, consequently, has not been extensively investigated in this study.

SELECTED SYSTEMS

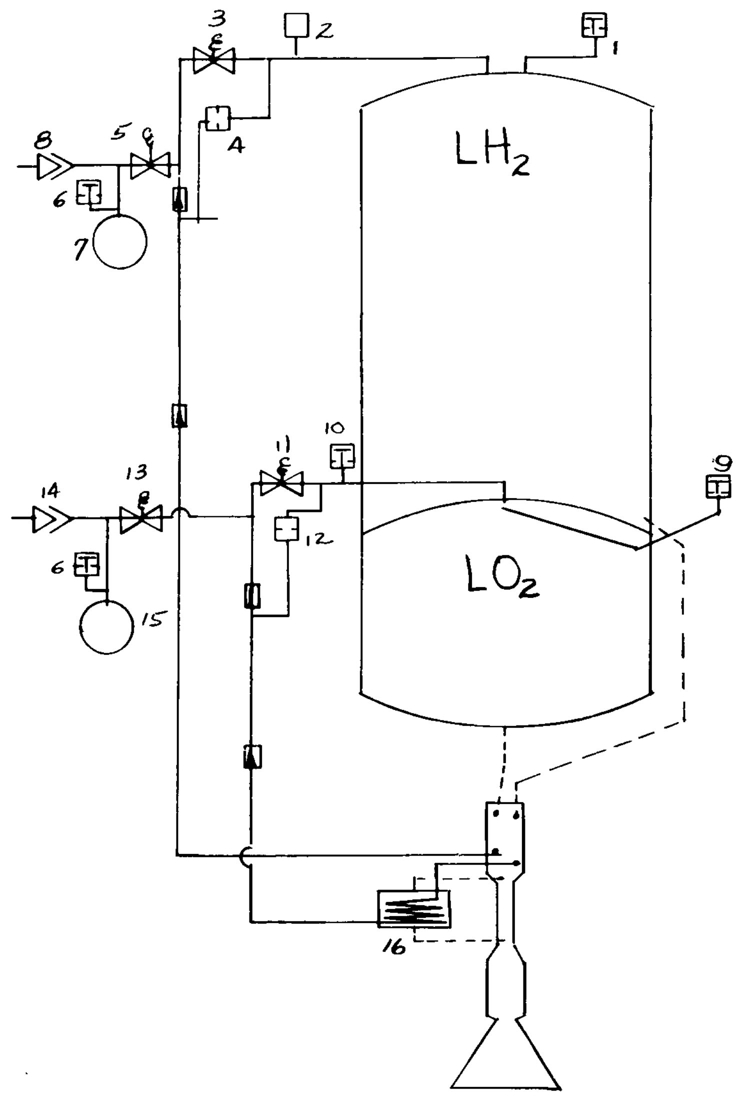
The main effort involving pressurizing system during this study has been directed toward a pressurization system in conjunction with a high pressure engine. It is felt that system optimization could be more readily accomplished with a high-pressure engine since the pressurization requirements could be incorporated into the design of the engine.

Fuel Tank Pressurization

LO_2/LH_2 engine fuel tanks will be pressurized by vaporized liquid hydrogen. Liquid hydrogen with a molecular weight of two is the lightest pressurant available. Its use for pressurizing the liquid hydrogen tank by tapping the hydrogen just prior to entry into the preburner in high-pressure engine, results in the lightest available system. Estimated hydrogen pressure at this point is 5000 PSI with temperature ranging from 300 R to 500 R. A schematic of a hydrogen tank pressurizing system is shown in Figure 10.

Liquid Oxygen Tank Pressurization

Pressurization of the LO_2 tank represents a compromise between system weight and reliability. The system recommended is a gaseous oxygen system, and its use is chosen for its inherent system reliability, its ready availability from the engine, and because it offers no extra handling problems.



LEGEND:

1. Vent Valve
2. Pressure Switch
3. Solenoid Valve - 1/4 inch
4. Orifice - 1/2 inch
5. Solenoid Valve
6. Relief Valve
7. Accumulator (2nd stage) 1.7 ft^3
8. Precharge Disconnect
9. Vent Valve
10. Pressure Switch
11. Solenoid Valve - 1/2 inch
12. Orifice - 3/4 inch
13. Solenoid Valve
14. Precharge Disconnect
15. Accumulator (2nd stage) 1/2 ft^3
16. GOX Heat Exchanger

Figure 10. Liquid Oxygen/Liquid Hydrogen Pressurization System

The only other pressurant that competes with gaseous oxygen and merits evaluation is helium as the main pressurant for LO_2 tank. Figure 9 shows the weight comparison of high-pressure stored helium pressurization system with that of gaseous oxygen pressurization system. Considerable weight saving can be realized by adopting a liquid helium system, but system complexity would increase. Figures 11 and 12 are schematic of the high-pressure storage helium system and liquid helium system, respectively. The main disadvantages of a helium system as compared to a gaseous oxygen system are scarcity of supply, lower system reliability, and handling and servicing requirements.

Sizing of a high-pressure gas pressurization system is always a problem. Inadvertent blow-down of the pressurant supply requires the use of large relief valves to prevent tank rupture. The quantity of stored gas for normal operations must also be based on some safety margin to assure propellant depletion prior to pressurant depletion.

Because of reliability and lack of system complexity, a gaseous oxygen system is recommended as a pressurant for the LO_2 tank. Oxygen bled from the compressor section would supply the source of pressurant. Hydrogen bled from the preburner section of the high-pressure engine, directed through a heat exchanger and returned to the mixing section would supply the necessary heat without any appreciable performance degradation. The LO_2 tank pressurization schematic is shown in Figure 10. Figure 13 depicts adaptability to the Pratt & Whitney high-pressure engine.

System Control Requirements

For the purposes of this study, constant ullage pressure sensing was used, and turbopump inlet sensing as an alternate control method was investigated. This method of pressure sensing imposes severe operational transients on the pressurization system because of the large variation in pressurant mass flow rates. The heat exchange necessary to satisfy pressurization requirements would undoubtedly become quite complicated. Components and lines sizes would become quite large to handle the tremendous flow rates required of the system at end boost due to loss of liquid heat.

Pressure Control

Control of ullage pressure flow to the propellant tanks has been shown by a solenoid valve in parallel with a fixed orifice. A pressure switch controls the solenoid valve. The choice of a solenoid valve in lieu of a conventional regulator was based on both reliability and performance. Although regulators appear to have greater potential reliability than a pressure switch-solenoid valve combination due to the absence of electrical requirements, equal reliability and performance have not yet been demonstrated.

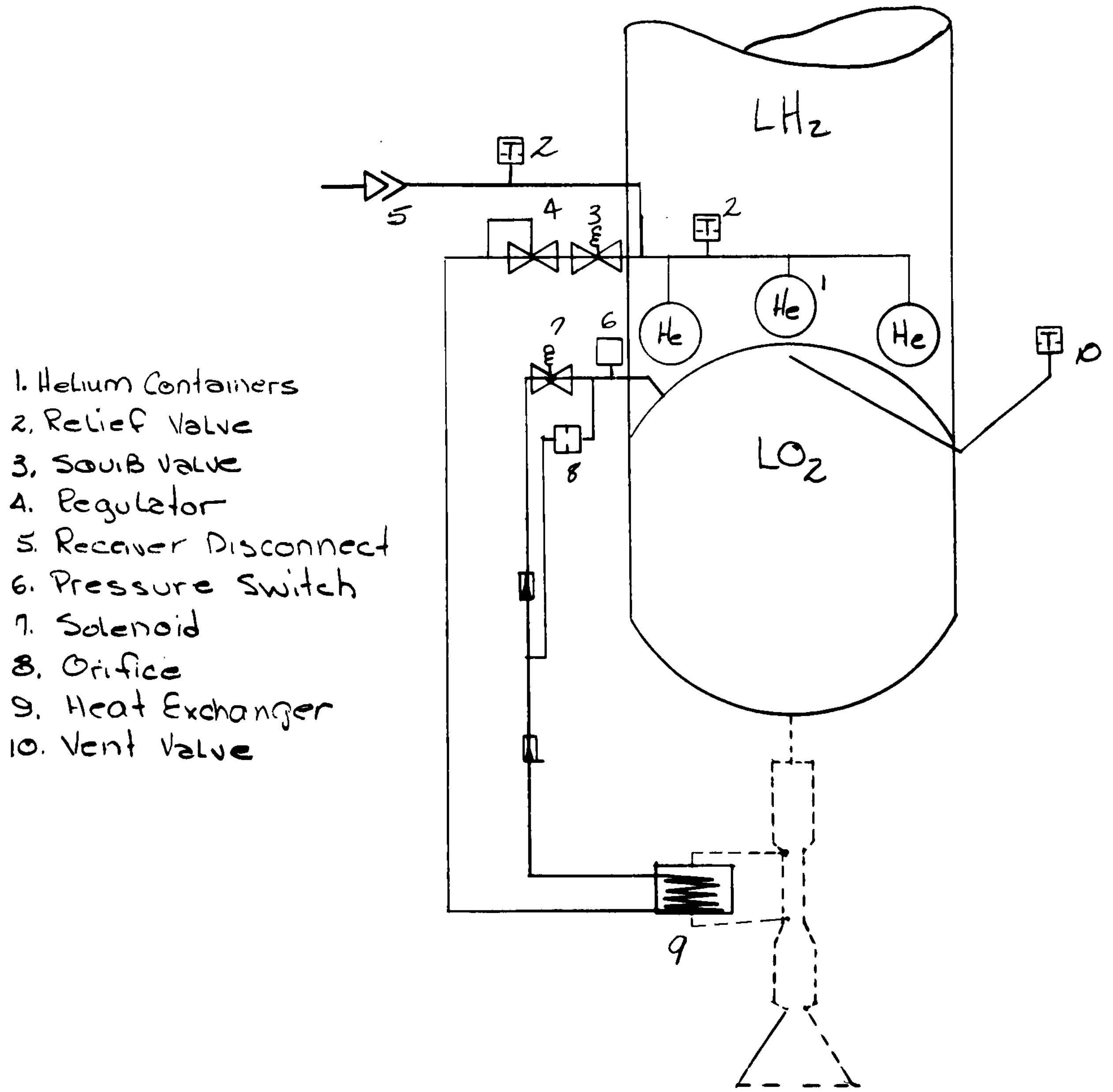


Figure 11. Gaseous Helium Pressurization System

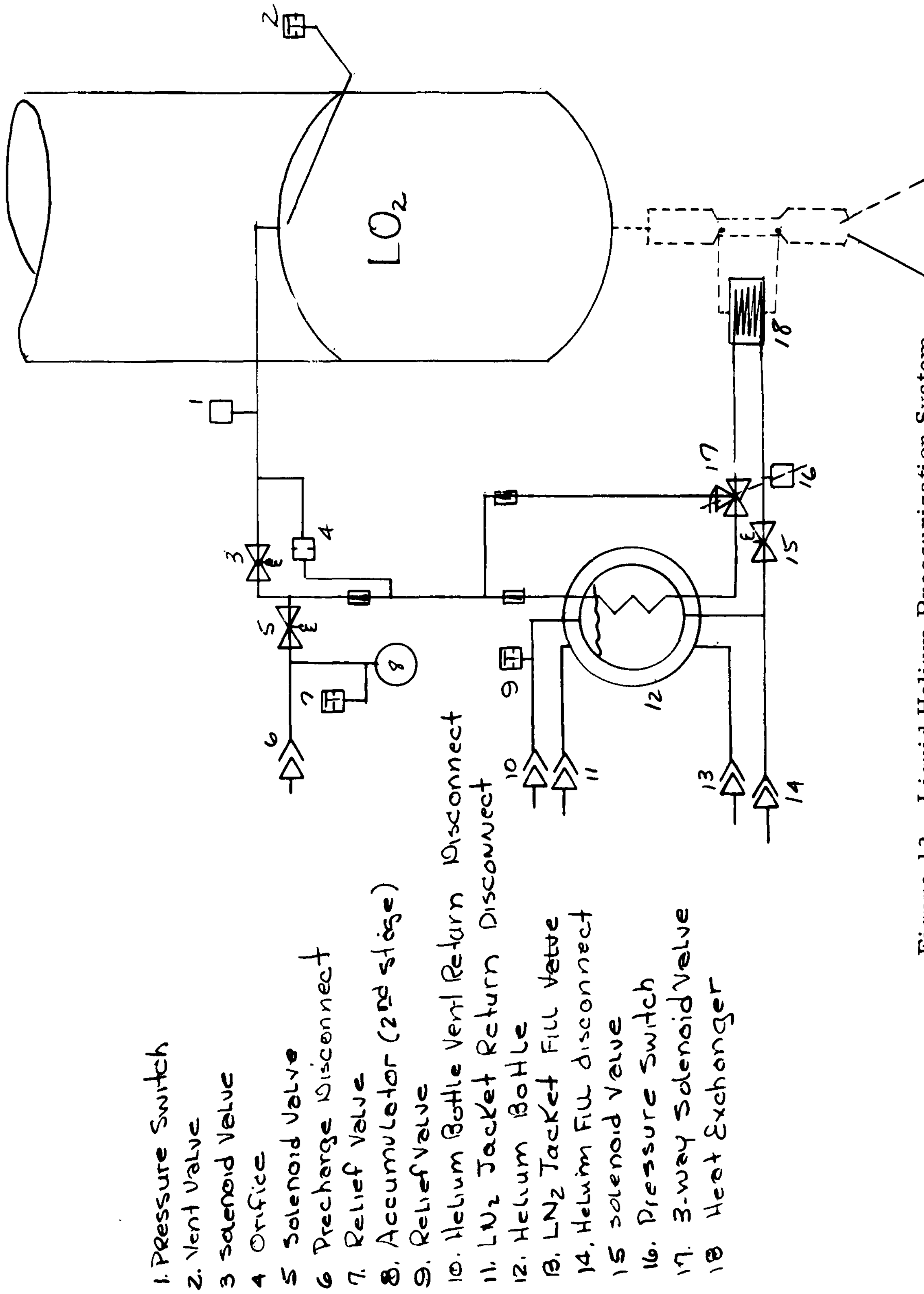


Figure 12. Liquid Helium Pressurization System

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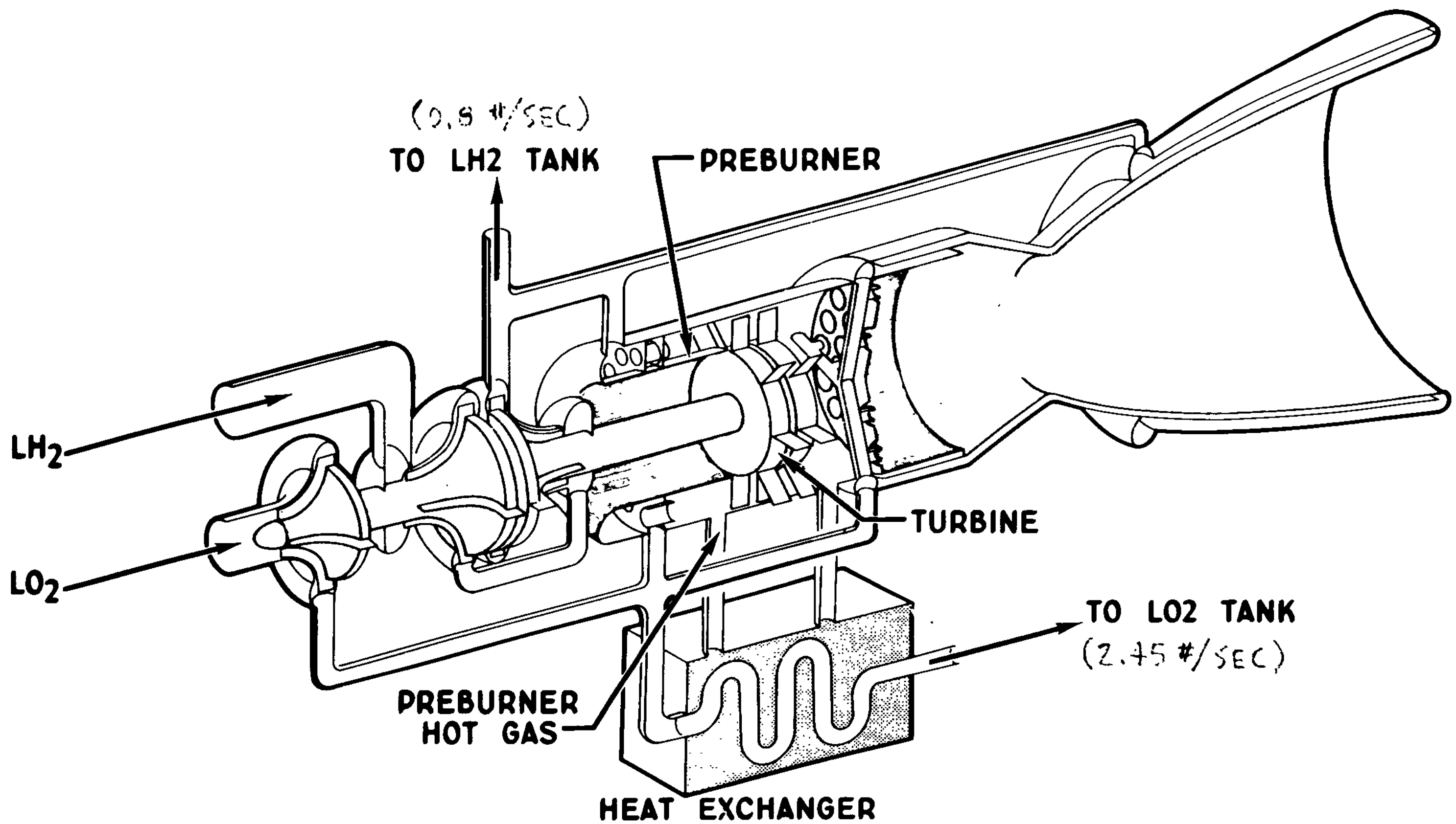


Figure 13. Liquid Oxygen/Liquid Hydrogen High-Pressure Engine Support Schematic

Clustered Engine

Engine clustering presents the choice of all engines contributing pressurant, or only a selected one or two. To prevent special requirements of any one engine, it is felt that each engine should provide a portion of the pressurization system. This, in turn, would be manifolded into a central feed line to the propellant tank. A failure of one pressurant source would not jeopardize the entire system.

SYSTEMS OPERATION REQUIREMENT

ATLITUDE CONTROL REQUIREMENTS

Throughout the boost trajectory, attitude control is required either by main engine gimbaling or an auxiliary control system. The capabilities of providing attitude control is necessary throughout the entire orbital trajectory; however, the magnitude of thrust required varies considerably. The initial boost requires thrust vector control to correct for engine misalignment, wind shear forces, and command attitude changes. The coast period requires a low thrust, low impulse system capable of maintaining the required orbital kick vector while the booster is essentially in a zero g environment. A slightly higher thrust level attitude system would be required during the kick phase to compensate for thrust vector misalignments and individual engines thrust variances.

Attitude control during the powered boost phase requires a considerably large moment to compensate for wind shear forces encountered relatively soon after launch. To compensate for wind shear, it is desirable to provide engine gimbaling capabilities. For a 600 K/300 K, LO_2/LH_2 booster, approximately 4 degrees gimbaling capability is necessary for sufficient booster control. Also, in the two-stage-to-orbit configuration, approximately the same gimbaling capabilities are necessary for the second-stage as for the first-stage wind shear requirements in order to control booster attitude during the separation phase of second stage. Roll moments resulting from engine misalignments are relatively small; and for a booster employing high-pressure engines, roll control could be accomplished by a hydrogen bleed from the engine turbine section to solenoid controlled jets. Estimates indicate approximately 12-40 pound thrust engines would be necessary to compensate for engine misalignments resulting in a roll moment, and 0.17 to 0.57 pounds-per-second mass flow demand would be required from the engine.

In a single-stage-to-orbit configuration, gimbaling of the engine becomes impractical due to the secondary expansion nozzle. An additional control system, preferably hypergolic, is needed to compensate for engine misalignments resulting in yaw or pitch moments. Attitude control requirements needed to compensate for wind shear could be accomplished by separate attitude control rockets or by the addition of surfaces and stablizing fins. The thrust of rocket engines required to compensate for wind shear effects is approximately 15,000 pounds of thrust if mounted forward of the LH_2 tank. The resulting moment would be equivalent to four degrees of engine gimbaling.

Engine misalignments could also be incorporated in the same system thereby eliminating an additional system. If surfaces and stabilizing fins were used in place of the rockets, engine misalignment moments from roll, pitch, and yaw would still require a separate rocket system to control the booster attitude at launch prior to the time the carnard surfaces become effective and at attitudes after these surfaces lose their control effectiveness. A weight comparison of the two systems indicate the surfaces incur a 2000 pound payload penalty over that incurred by a rocket system.

During the booster coast period, attitude control is necessary for guidance gyros stabilization. Also, prior to initiating the propulsion kick-into-orbit, alignment of the vehicle for the kick vector is necessary. Since, during this period, the vehicle is in a zero g environment, total impulse requirements for maintaining attitude are small. A potential source of energy which could be made available for attitude control during this period are ullage gases. Venting of these gases through strategically located jets could suffice as the coast period attitude control propulsion system. However, with a multiple solid-engine kick system an additional control requirement is necessary during the orbital kick-thrust period to compensate for engine misalignment or thrust differentials between engines. Thrust requirements necessary to maintain payload vector are considerably higher than those required for the coast period; consequently, jets operated on ullage gases could not be used for this application because of the limited thrust level (10 pounds). A separate system would be necessary to meet the orbital kick control requirements in this case. In order to eliminate added components and overall complexities such as valves and power supply, the orbital kick control system would take care of coast period requirements with very little added weight, as in the additional control jets and power supply required of an ullage gas system.

ORBITAL KICK REQUIREMENTS

When the booster plus payload reaches apogee after an approximate coast period of 45 minutes, an impulse must be provided to place the vehicle into a circular orbit. The nominal impulse necessary would be that which imparts a nominal velocity increment of approximately 395 feet per second to the vehicle or payload. Methods of obtaining the necessary impulse are utilization of booster engine and main propellant tank, utilization of booster engine or separate kick engine with positive expulsion, or by use of solid engines.

One of the main problems associated with liquid-engine start for orbital kick purposes is the zero g environment. In a liquid-engine system ullage rockets are normally required to position the main tank propellants at the pump inlet prior to main engine start. Propellant positioning can also be accomplished by means of positive expulsion where propellants are retained in a container throughout the zero g flight and expelled to the pumps by a bladder at the time of engine-start.

Utilization of booster engine and main propellants to provide the necessary impulse presents problems in maintaining the cryogenic propellants and components in a usable condition acceptable for engine operation. Procedures must be provided which will insure the availability of LO_2 and LH_2 to the turbopumps at minimum NPSH requirements. Since initial acceleration supplied by ullage rockets is low, the pressurization system must provide the necessary pressure to satisfy the engine pumps.

During the zero g environment the action of the fluid within the propellant tank is subject to considerable conjecture. One school of thought is that fluid will spread uniformly over the inside surfaces during zero g conditions; another is that the fluid will eventually form as an elliptical fog in the middle of the tank. In either case, boil-off, or bulk temperature rise, will occur due to the transfer of heat from the tank walls, which are heated aerodynamically during the boost and by solar radiation during coast. During the propellant positioning phase prior to orbital ejection, an additional heat input is imparted to the liquid resulting in a bulk temperature rise or boil-off. To provide satisfactory turbopump NPSH without prohibited increase in ullage pressure, the liquid should be subcooled by lowering tank pressure to permit additional boil-off until the temperature is at the desired level for engine operation.

To accomplish this requirement, a complex pressurizing and vent system becomes necessary. Additional vent valves to handle the boil-off during the cool-down phase would be required. Excessively large amounts of stored gas are required for tank pressurization since ullage pressure makeup is necessary after the liquid subcooling period. Chill-down of engine pumps and components to operating temperatures may require as much as 3000 pounds of propellants after the engine-inoperative period of 40 to 45 minutes.

A positive expulsion, bladder-type feed system in conjunction with a separate pressure fed LO_2/LH_2 engine to eliminate chill-down would be another method of providing the kick impulse. The main disadvantage of such a system is the lack of reliable expulsion bladders for cryogenic propellants. The use of storable, hypergolic propellants with positive expulsion is also feasible. System complexities are reduced mainly because of the elimination of a spark ignition system.

Present method to accomplish kick impulse is the use of solid rocket engines. Reliability of solids are high and have excellent control of total impulse. Although specific impulse is low compared to liquids, the absence of supporting systems results in lower weight and higher reliability. A comparison of liquid and solid kick systems are shown in Table 9.

Table 9. Kick Engine Weight Comparison For 600 K Single-Stage-To-Orbit

System	Weight		Remarks
Main Propulsion			
Propellants	1,350		Low Reliability
Evaporation	1,500		
Pressurization	700		
Chill Down	* 3,000		
Total		6,550	
Solid Engine			
Engine Propellant		2,550	High Reliability
Pressure-Fed LO_2/LH_2			
Engine	185		Low Reliability
Pressurization and Tanks	365		
Propellants	1,860		
Total		2,410	
Pressure-Fed N_2O_4/UDMH			
Engine	186		Good Reliability
Pressurization and Tankage	100		
Propellants	2,200		
Total		2,486	

*Chill-down estimate is a maximum. Further information is necessary on pump temperature sensitivity for a more accurate propellant weight requirement.

CONCLUSIONS

There is no obvious cost advantage with the advanced engine systems, although the high-pressure engine is competitive in most all cases. The best argument for the high-pressure engine appears to be with the single-stage-to-orbit configurations. The estimates presented for this vehicle are probably optimistic due to lack of accurate performance and detail design data for an engine system utilizing secondary expansion. The high-pressure engine concept appears feasible; but the desirability of such a system may depend upon the degree of success realized with advanced systems, such as the nuclear rocket engine presently under active development. Similarly, plug-nozzle and solid-propellant engines do not present definite cost advantages when used with the two-stage vehicles considered during this study.

With the possibility of development of a high-pressure engine, considerations should be given to the type and method of propellant tank pressurization which will be associated with the vehicle. A vaporized propellant pressure system has definite advantages over other methods of pressurizing. The ability of the high-pressure engine to support such a system is considered mandatory.

Present high-pressure engine design information reveals that the use of boost inducers are necessary to provide the required turbopump NPSH without incurring undue weight penalty because of high ullage pressure requirements. Preliminary studies indicate if the NPSH requirements of the boost inducer system are maintained relatively near the propellant vapor pressure, the resulting weight savings due to reduced ullage pressure more than compensates for the boost inducer system weight.

FLIGHT MECHANICS

SUMMARY

The most promising design philosophies and vehicle configurations have been selected for this Cost Optimized Booster System Study. This section summarizes the flight mechanics investigations which were conducted to satisfy the study requirements.

The initial portion of the studies considered first-stage thrust levels of 600K, 800K, 1500K, and 3000K for two-stage boosters with single tankage systems for each stage. The effects on staging optimization of a Pratt & Whitney high-pressure engine and conventional first-stage engine configurations with pump-fed bell and plug nozzles were studied. Also evaluated were the effects of upper stage conventional and high pressure engines. All liquid LO_2/LH_2 propellants were analyzed in both stages; LO_2/RP was considered in all first-stage applications except those utilizing the high-pressure engine cycle. All configurations were compared on a basis of dollars per pound of payload in a 300 nautical mile orbit.

Based upon selected optimum systems of the above, the final portion of the flight mechanics studies considered:

1. Two-stage boosters utilizing clustered modules in the first stage and a common module in the second stage.
2. Two-stage boosters utilizing clustered modules in the first stage and a staging optimized single tankage second stage.
3. Two-stage boosters utilizing a segmented solid for the first-stage propellant.
4. Two-stage boosters utilizing identical nozzle expansion ratios for both stages.
5. Two-stage boosters utilizing different chamber pressures in the first-stage Pratt & Whitney advanced high-pressure engine cycle.
6. Two-stage boosters utilizing a liquid fluorine system (LF_2/LH_2) as the second-stage propellant.

7. A single-stage booster utilizing secondary expansion to achieve greater payload capability.

8. Two-stage boosters utilizing recovery of the first stage at the optimum staging point and at various extended staging points.

All of the above concepts have been evaluated and the results are discussed in this section of the report.

The performance evaluations and optimization studies have been supplemental with peripheral data on dynamic considerations, flight control, and recovery analysis with special application to large booster systems.

PERFORMANCE ANALYSIS

METHODOLOGY

The selection of the most promising design philosophies and vehicle configurations for this study has been made on the basis of a booster system cost optimization. This comparison basis of dollars per pound of payload in orbit has included the effects of flight mechanics, propulsion systems, vehicle design, structure, weight, reliability, and economics and operations. This section presents an outline of the staging optimization analysis of flight mechanics which was conducted in order to satisfy the study requirements.

In the course of this study program, primary emphasis was placed on a detailed parametric performance analysis of the boost systems as described in Section V - Appendix. This parametric analysis complied with the ground rules specified in Section III of this report.

An IBM 7090 booster design program has been developed for estimating the performance of multi-staged boost vehicles for various earth, lunar, and space missions. The program, which is shown schematically in Figure 14, includes the effects of vehicle drag, gravity losses, rotating earth, and the variation in thrust, specific impulse, and gravitational attraction with altitude. The variation in booster mass fraction, ν (ratio of propellant weight to booster weight) with booster size and thrust level is also taken into account.

In order to minimize the machine time required for large parametric performance studies, the program described was based on a ballistic boost path for the two stages. A ballistic path does not represent the optimum path and will not result in the maximum payload capability of a given boost system; however, past studies performed at the Space and Information Systems Division have shown that the ballistic path will result in near maximum orbited payloads as long as the second-stage initial thrust-to-weight ratio, $(T/W_o)_2$, is greater than 1. For example, an optimum trajectory calculation for a typical Saturn configuration where $(T/W_o)_2$ was approximately 1.6 yielded an increase in payload capability of less than 3 percent over a pure ballistic path. Therefore, the payload trends and magnitudes based on a ballistic path should be valid for this study since in the majority of the cases $(T/W_o)_2$ is greater than 1. In those few cases where $(T/W_o)_2$ is less than 1, a certain degree of caution should be exercised in accepting the orbited payload values as near optimum.

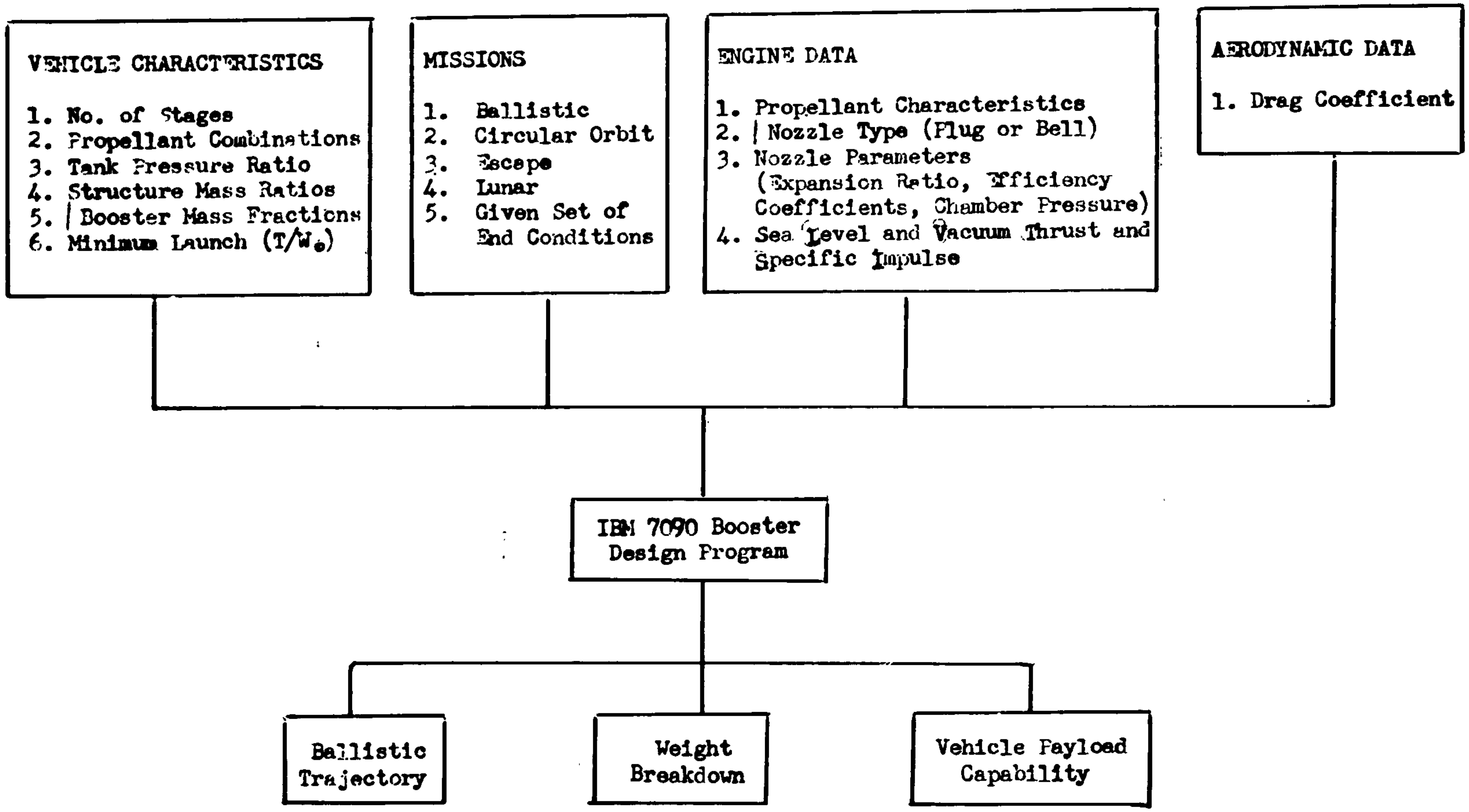


Figure 14. Booster Design Program

Unfortunately, time did not permit a re-evaluation of these few cases. The effects of drag variation on large vehicle performance is small. At no time did the drag losses exceed 10 percent of the total losses; thus, a standard value for the vehicle drag parameter was used in the program without introducing a significant error. The vehicles have been sized for 3 percent excess velocity to account for steering losses and variations in thrust, specific impulse, and drag.

For the particular mission studied, namely a two-stage vehicle boosted into a 300 nautical mile circular orbit, a variety of operational modes can be achieved. For this study the orbital mission was achieved assuming 180 degrees coast-to-apogee following the thrust cut-off with a kick-into-apogee to impart the orbital velocity. It may be that this mission profile is undesirable, but the assumptions so made permitted straightforward computations with realistic results. The simplifications in computation achieved by the described assumptions resulted in the production of much larger volumes of data within the scope of the study than would have been possible otherwise.

TWO-STAGE BOOSTER SYSTEMS

Based on given thrust levels, engine types, and propellant combinations per stage, Figures 15 thru 22 show the variation in mission payload (W_{PL}) versus first-stage velocity increment (ΔV_1) as a function of second-stage initial thrust-to-weight ratio, $(T/W_o)_2$. The performance analysis ground rules applicable to this section are presented below:

1. $(T/W_o)_1 = 1.20$
2. $T_1 = 600K$ and 1500K. Two 800 K systems and two 3000K systems will be established predicated on the most promising 600K and 1500K systems respectively. These will employ (a) state-of-the-art propulsion systems and (b) advanced propulsion systems.
3. Two-stage booster systems with single tankage for each stage will be employed.

600K Systems

Figures 15 and 16 present the staging optimization results for all the 600K systems considered. The effect of various first-stage engine and propellant types with a (0.2) JH upper stage is shown in Figure 15. The PH/JH configuration has a slight payload advantage (30,400 pound) over that of a WH/JH configuration (30,200 pound) Figure 16 shows the effect of substituting a WH upper stage in the place of the JH system. Since the WH system is still a paper design, it has no specific thrust level.

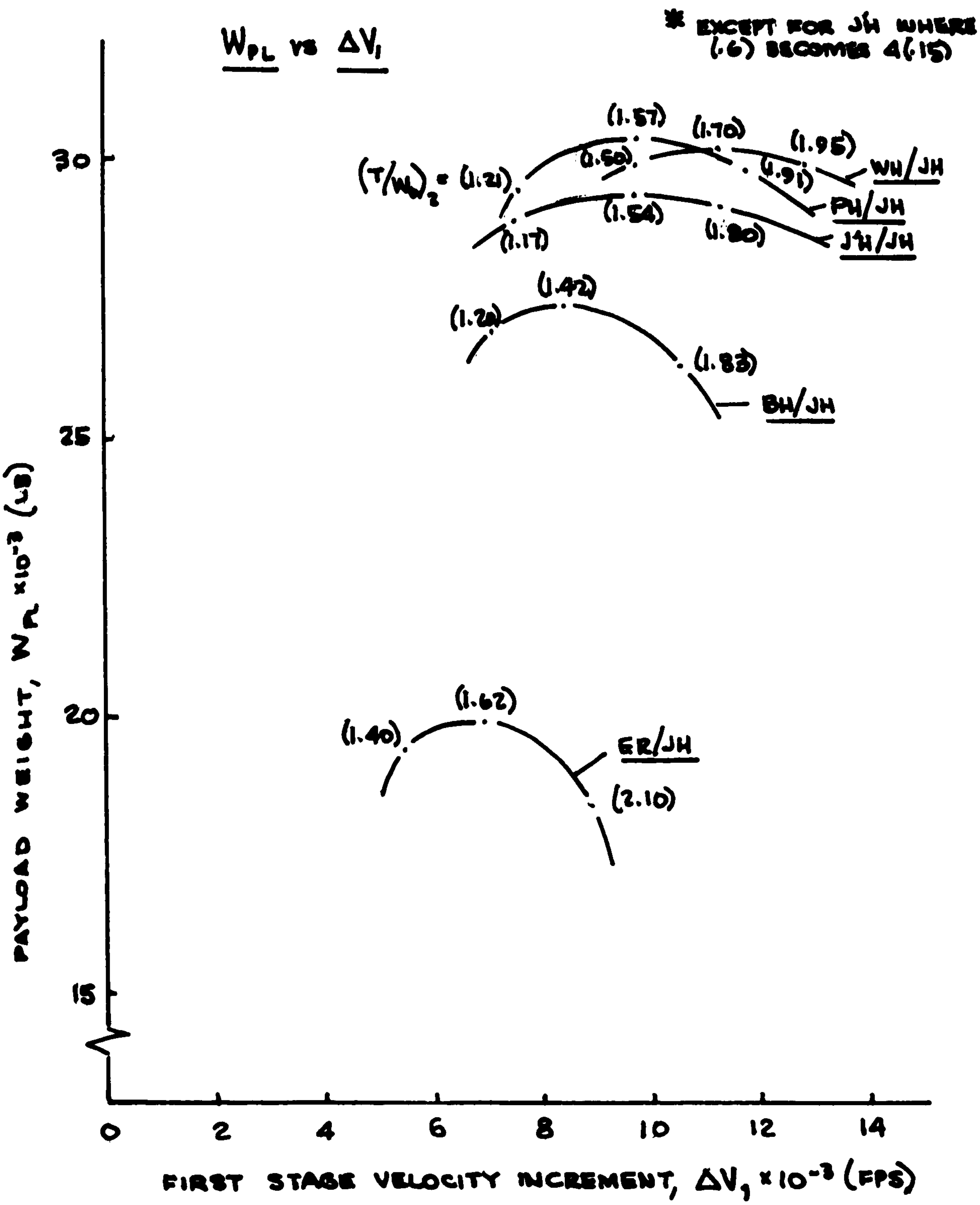


Figure 15. Payload Weight Versus First-Stage Velocity Increment for 0.6/0.2 JH

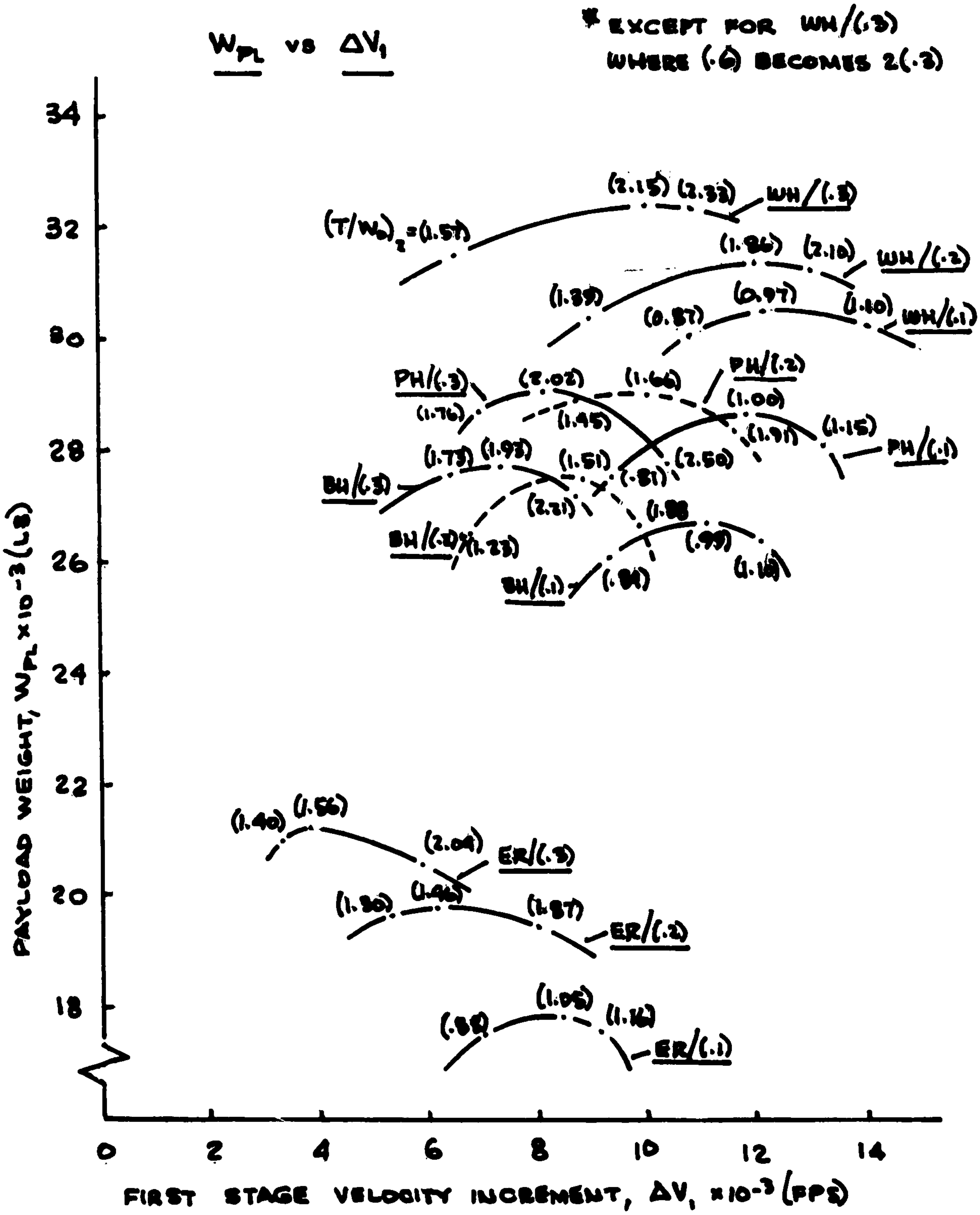


Figure 16. Payload Weight Versus First-Stage Velocity Increment for 0.6/W'H

Therefore, three upper-stage thrust levels were selected (T_2 = 100K, 200K, 300K) and analyzed. One further ramification was made; by using a lower stage two engine installation it was possible to forsee a cost reduction in the WH/(0.3) system. That is, a 2(0.3) WH/(0.3) W'H vehicle was analyzed in place of a (0.6) WH/(0.3) W'H design. Figure 16 shows a distinct payload advantage for this 2(0.3) WH/(0.3) W'H system.

It should be noted for this system and for later discussed systems that vehicles where RP fuel was used in the lower stages and LH_2 fuel used in the upper stages, the staging optimization favored the LH_2 stages for propellant mass fractions. The optimum ΔV_1 for these 600K systems ranged from 4000-8000 fps, whereas for an all LO_2-LH_2 (lower stage)/LO_2-LH_2 (upper stage) system the ΔV_1 ranged from 8000-14,000 fps. Predicated upon the type of second-stage engine, the optimum performance $(T/W_o)_2$ for the W'H concept was 2.15 and for the JH concept was 1.57. At this point the cost comparison was made of all the 600K systems; the optimum turned out to be the 2(0.3) WH/(0.3) W'H vehicle. This selection is valid as long as it is remembered that it was made on a relative basis and that its sole purpose was to eliminate all but the cost optimized design. To arrive at the selected system's true optimized cost an iteration procedure is involved. That this is so is seen when it is remembered that the engineering analyses of the propulsion, system weights, and guidance have all been done separately. Each group of equations is sufficiently complicated that it is impossible to make solutions for so many configurations in closed form, and still stay within the scope of the study. There is a very close coupling between the various analyses; so a change in structure weight affects the propellant weights, which in turn changes the payload weight, which changes the structure weight. This coupling arises because the outputs of each analysis serve as inputs for the others. The results of performing two iterations is shown in Figure 17. The payload weight has increased from an initial maximum of 32,500 pounds to a final maximum of 40,500 pounds.

1500K Systems

Figures 18, 19, and 20 show payload capabilities for all the 1500K systems. Figure 18 shows the effect of various first stages with one-, two-, three-, or four- engined JH upper stages. The 4(0.375) WH/2(0.2) JH vehicle has the maximum payload capability (W_{PL} = 88,000 pounds. By substituting a PH upper stage for the JH and considering second stage thrust levels of 300K, 400K, and 500K, Figure 19 indicates the 4(0.375) WH/(0.3) PH is the best performer (W_{PL} = 80,000 pounds. With a W'H serving as the upper stage, Figure 20 shows that the maximum payload (89,000 pounds is orbited by a 2(0.75) WH/(0.75) W'H configuration. A dual first-stage engine was rationalized in the same manner as the

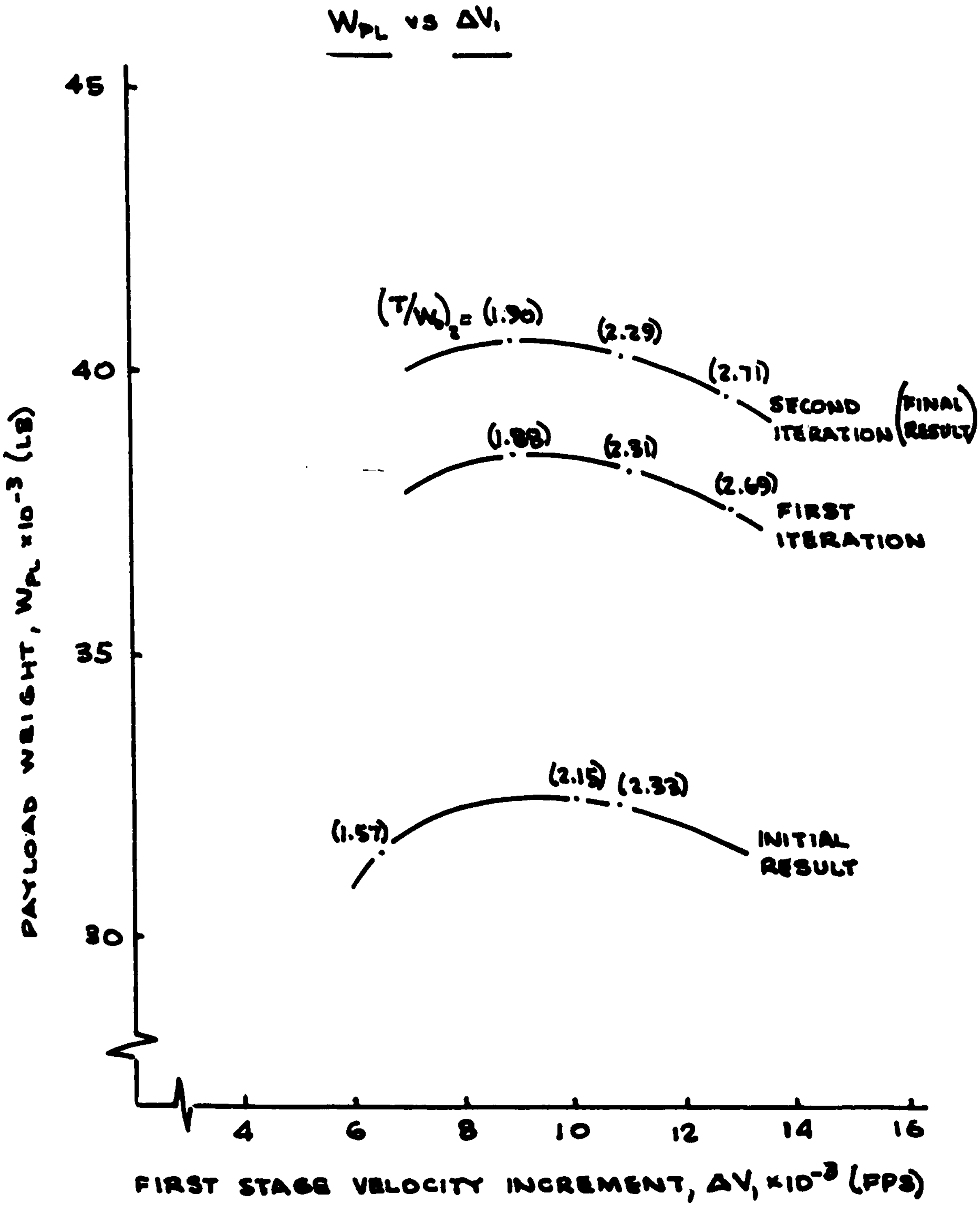


Figure 17. Payload Weight Versus First-Stage Velocity Increment for 2(0.3) WH/0.3 W'H Selected System

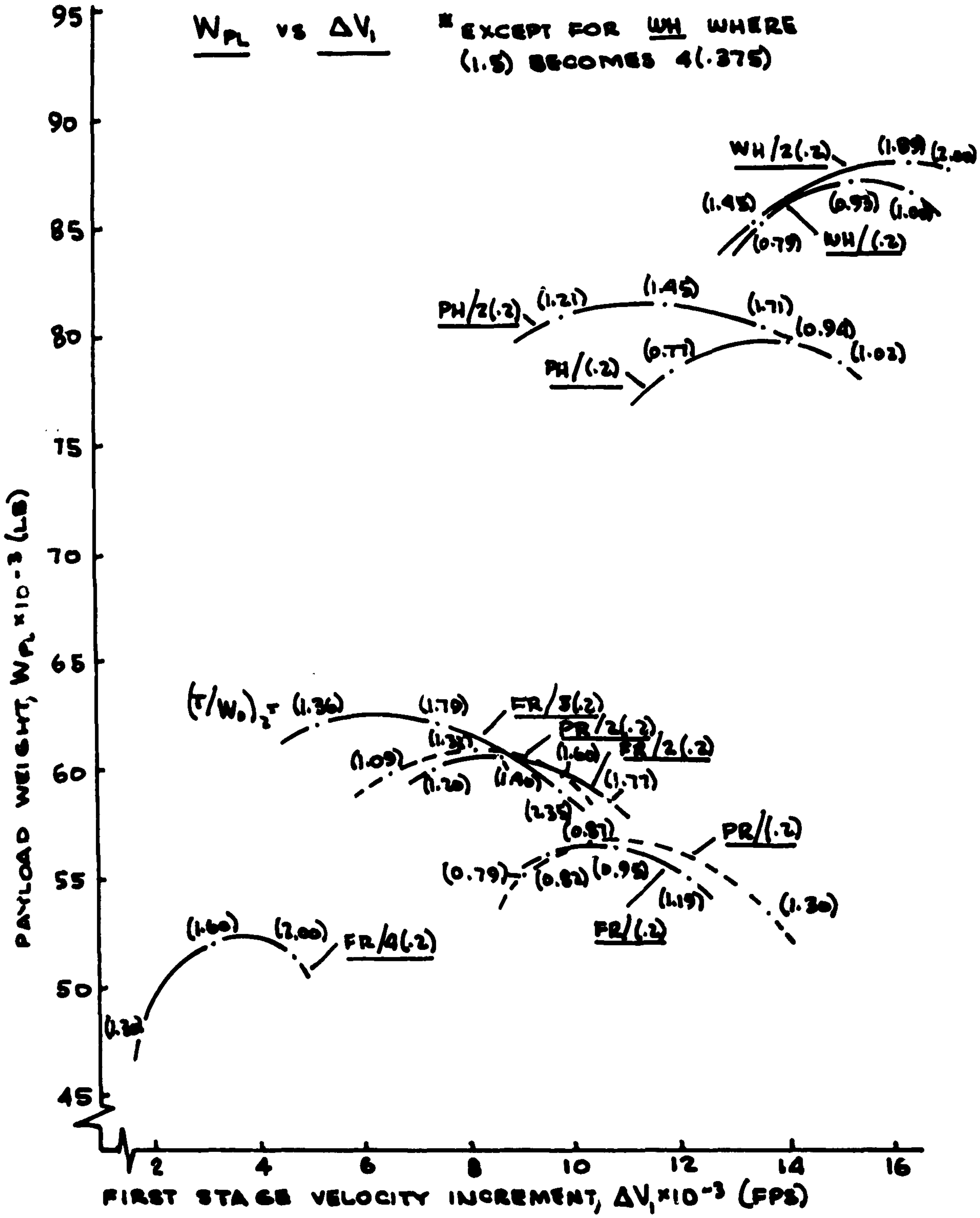


Figure 18. Payload Weight Versus First-Stage Velocity Increment for 1.5/JH

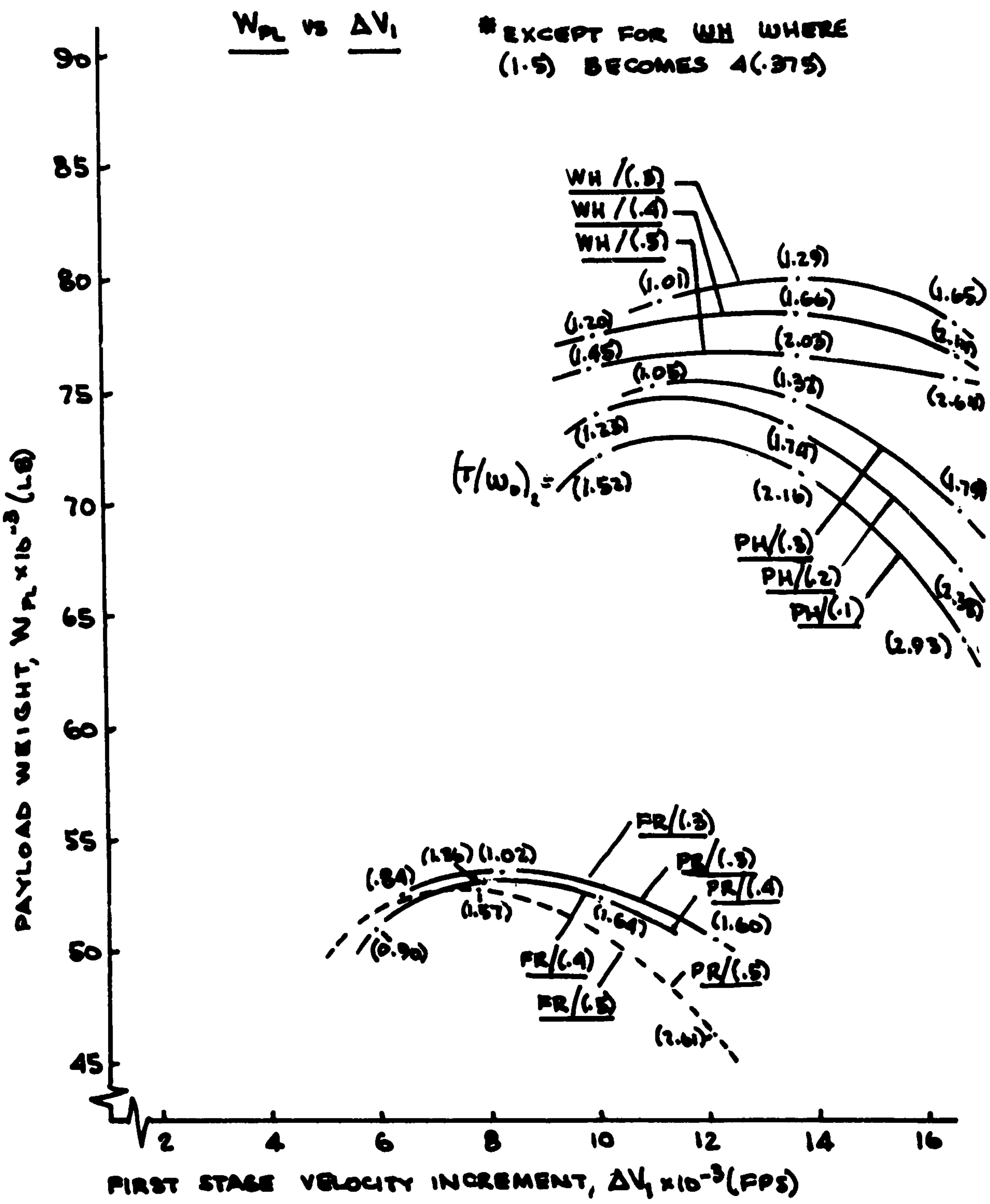


Figure 19. Payload Weight Versus First-Stage Velocity Increment for 1.5/PH

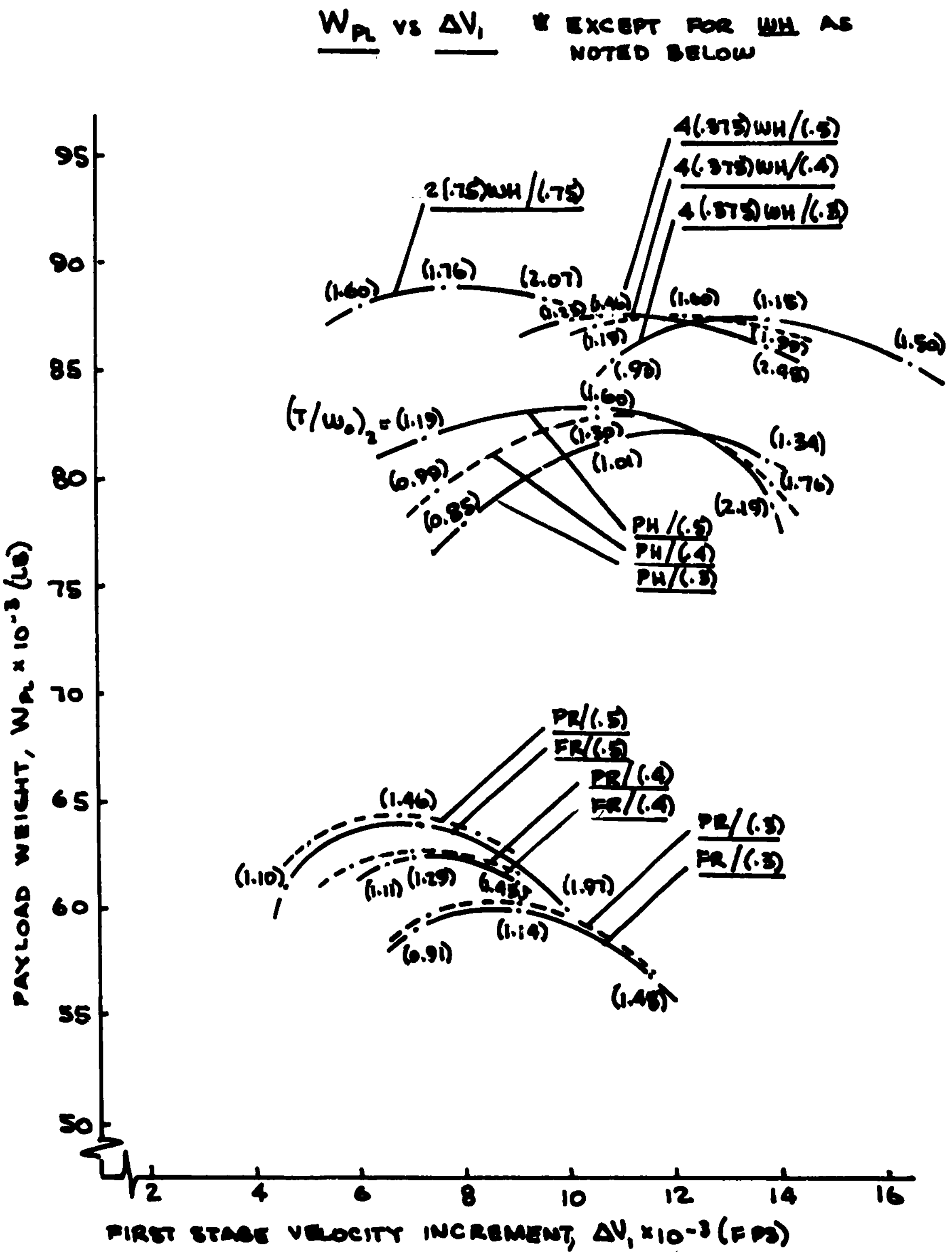


Figure 20. Payload Weight Versus First-Stage Velocity Increment for 1.5/W'H

600K system. Based on the type of second-stage engine, the performance optimum $(T/W_0)_2$ for the JH concept was 1.89, for the PH concept 1.29, and for the W'H concept 1.76. After a cost analysis, the 2(0.75) WH (0.75) W'H system was selected as the cost optimum of all the 1500K systems.

800K Systems

With the 2(0.3) W'H vehicle established as the optimum 600K system, an 800K selection is greatly simplified. Because the 600K system is an advanced propulsion system, a state-of-the-art propulsion system must be concurrently analyzed as stated in the ground rules. Once again a dual engine system was considered, 2(0.4) WH/(0.4) W'H, for the advanced propulsion system. For the state-of-the-art propulsion system two configurations were selected because it was not known beforehand which of the two had the better payload capability, 5(0.6) J''H/(0.2) JH and 5(0.16) J''H/2(0.2) JH. The results are presented in Figure 21. Once again the WH/W'H combination demonstrated its superior payload capability by orbiting 52,800 pounds, 8000 pounds more than its competitor, 5(0.16) J''H/(0.2) JH.

3000K Systems

In a manner similar to that used for 800K systems, the advanced 4(0.75) WH/2(0.75) W'H system was compared with two state-of-the-art systems, 2(1.5) FR/4(0.2) JH and 2(1.5) FR/3(0.2) JH. The WH/W'H outperformed the 2(1.5) FR/4(0.2) JH vehicle by 65,800 pounds by putting into orbit 188,400 pounds as compared to 122,600 pounds. The results are shown in Figure 22.

ALTERNATE BOOSTER SYSTEMS

With the selection of the 2(0.3) WH/(0.3) W'H and 2(0.75) WH/(0.75) W'H vehicles as the cost optimized 600K and 1500K two-stage booster systems respectively, various alternate booster systems have been considered and analyzed. Because of limited time for study, the performance capabilities of a great many of the 1500K alternate booster systems have not been evaluated. However, a complete performance analysis has been made of the 600K alternate booster systems and is discussed in this section.

Clustered Module - Common Modules for First and Second Stage

Three 600K modular systems based upon the most promising first-stage single-tank LO_2-LH_2 system, 2(0.3) WH, have been established. These systems have a three-, four-, and seven- module arrangement for the first stage with a similar single module for the second stage. The particular common modules considered were 3 [2(0.3) WH] /2(0.3)WH;

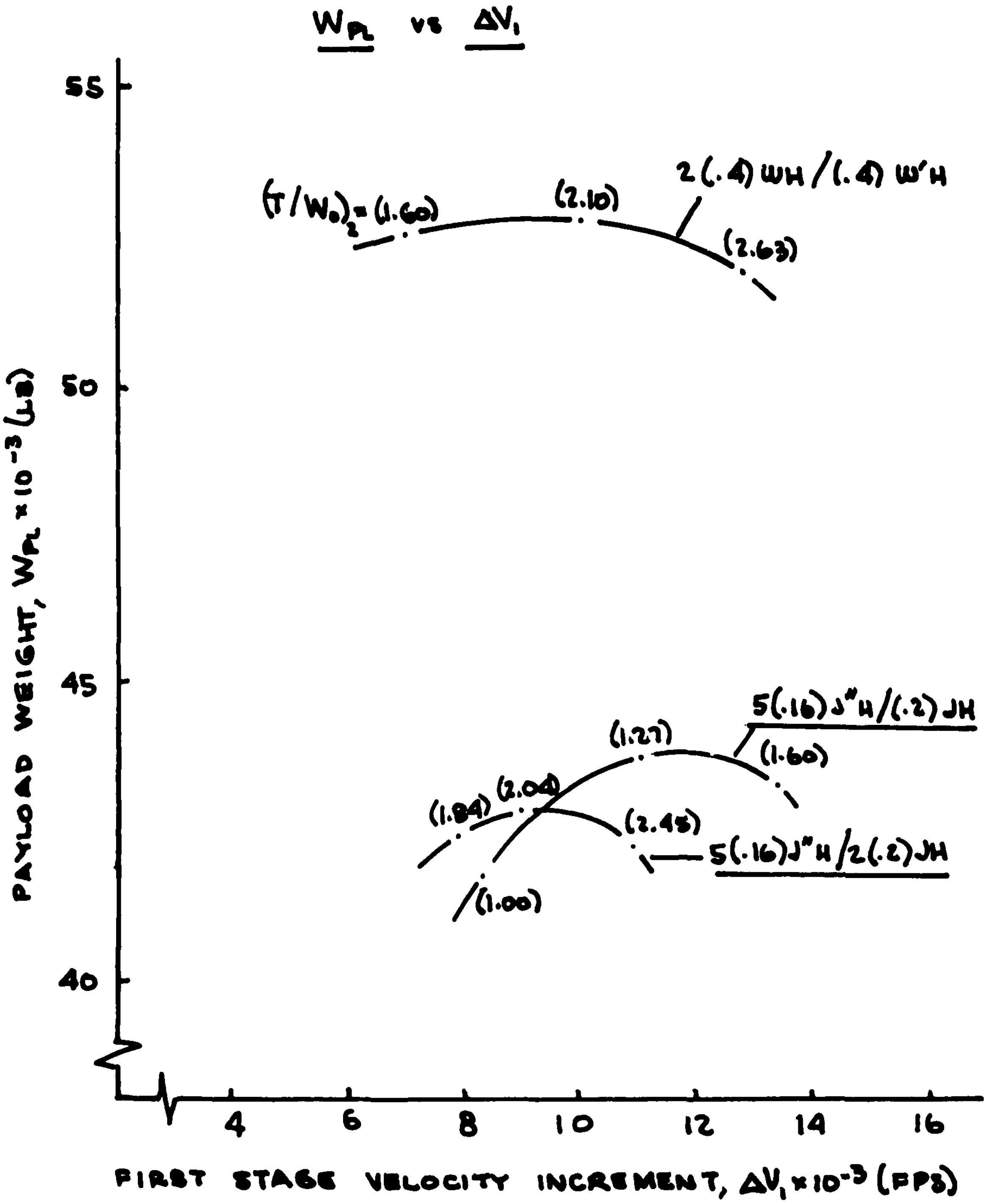


Figure 21. Payload Weight Versus First-Stage Velocity Increment for Selected 800K Systems

$\underline{W_{PL}}$ vs $\underline{\Delta V_1}$

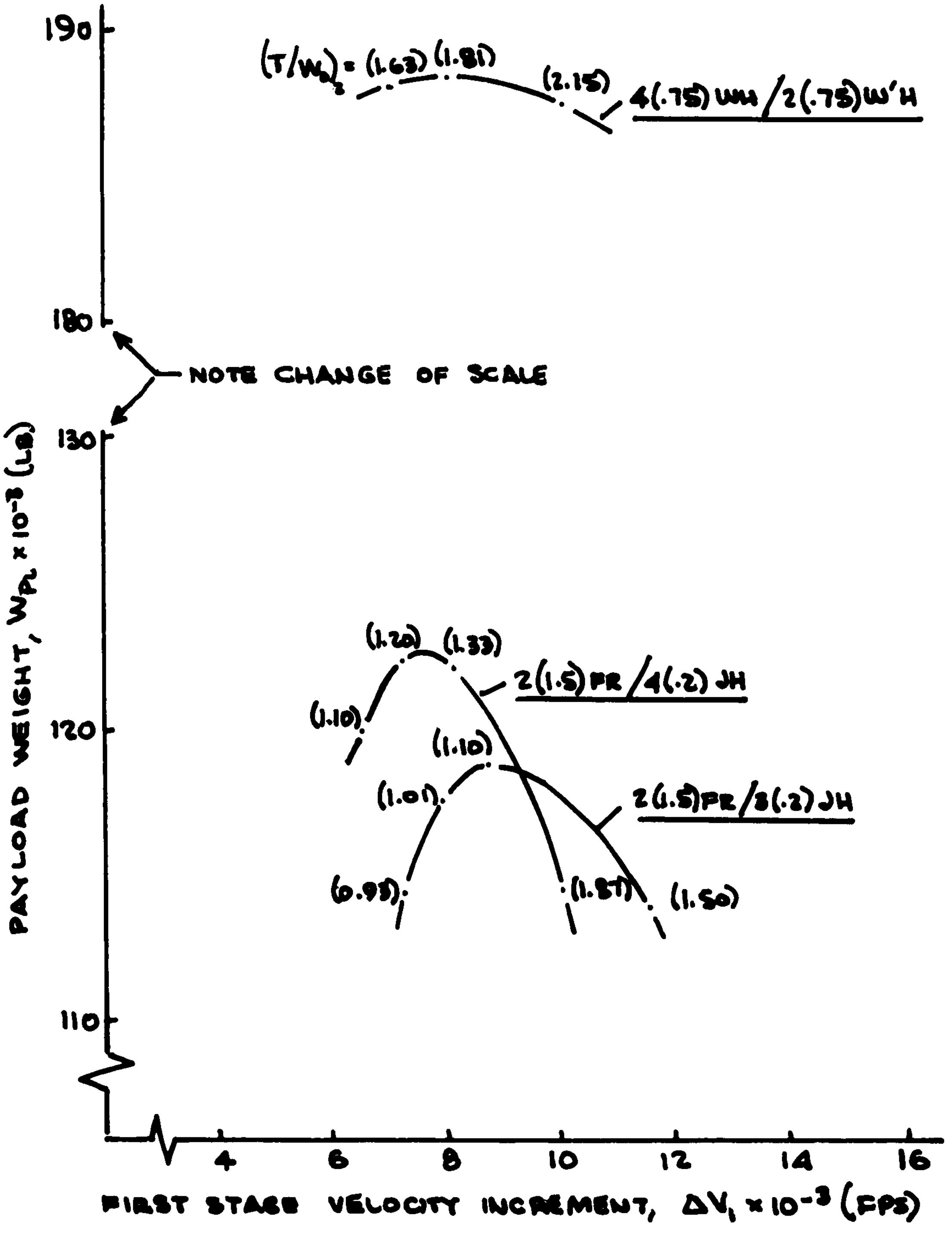


Figure 22. Payload Weight Versus First-Stage Velocity Increment for Selected 3000K Systems

4 [2(0.3) WH]/2(0.3) WH; and 7 [2(0.3) WH]/ 2(0.3) WH.

The ground rules followed for the common-module concept are discussed below:

1. $(T/W_o)_{1\,min} = 1.20$. In the Two-Stage Booster Systems Section of this study, the initial thrust-to-weight ratio was set as exactly equal to 1.20. For the modular concept, the ground rule is that the $(T/W_o)_1$ can assume any value so long as it does not become less than 1.20.

2. The propellant weight per module can be equal to or less than 330,000 pounds, but cannot exceed this weight. This value is equal to the propellant weight of the first stage as determined for the 600K selected two-stage booster system. The three-module arrangement cannot, therefore, have more than 990,000 pounds of propellant in the first stage and no more than 330,000 pounds of propellant in the second stage. Similarly, the four-module concept cannot have more than 1,320,000 pounds of propellant in the first stage and no more than 330,000 pounds of propellant in the second stage. The same formulation is applied to the seven-module system.

The results of the three-common-module arrangement are shown in Figures 23 through 25. Shown in Figure 23 is payload weight (W_{PL}) versus first-stage propellant weight (W_{F_1}) for various values of second-stage propellant mass fraction (υ_2). Also shown is the 990,000 pound constraint for W_{F_1}; anything greater than 990,000 pounds is unacceptable. Figure 24 presents the same payload weight versus second-stage propellant weight (W_{F_2}), again for various values of υ_2. The second-stage propellant constraint (W_{F_2} = 330,000 pounds) is shown; anything larger is unacceptable. The data shown in Figure 25 are obtained by a crossplot of Figures 24 and 25 and are plotted versus first-stage velocity increment, ΔV_1. The three constraints, W_{F_1}, W_{F_2}, and $(T/W_o)_1$ are shown. The results indicate, without violation of any of the constraints, that the maximum payload orbited for the common three-module arrangement is 105,400 pounds. It also has the following properties:

1. $(T/W_o)_1 = 1.20$

2. ΔV_1 = 9500 fps

3. W_{F_1} = 990,000 pounds (capacity).

The second-stage propellant weight is established by noting in Figure 23 that the locus for $(T/W_o)_1 = 1.20$ and W_{F_1} = 990,000 pounds is $\upsilon_2 = 0.70$

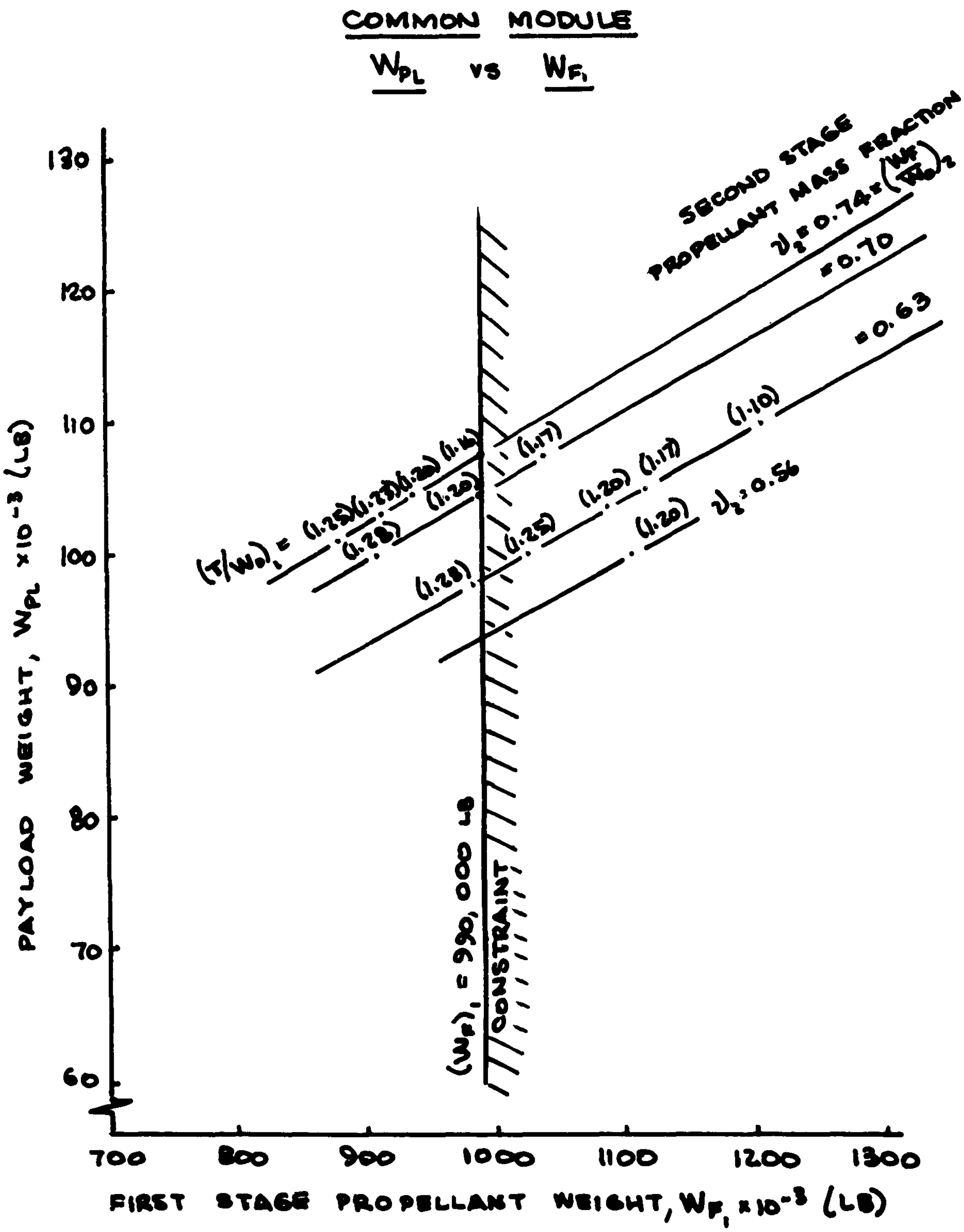


Figure 23. Payload Weight Versus First-Stage Propellant Weight for 3 [2(0.3)WH] / 2(0.3)WH Common Module

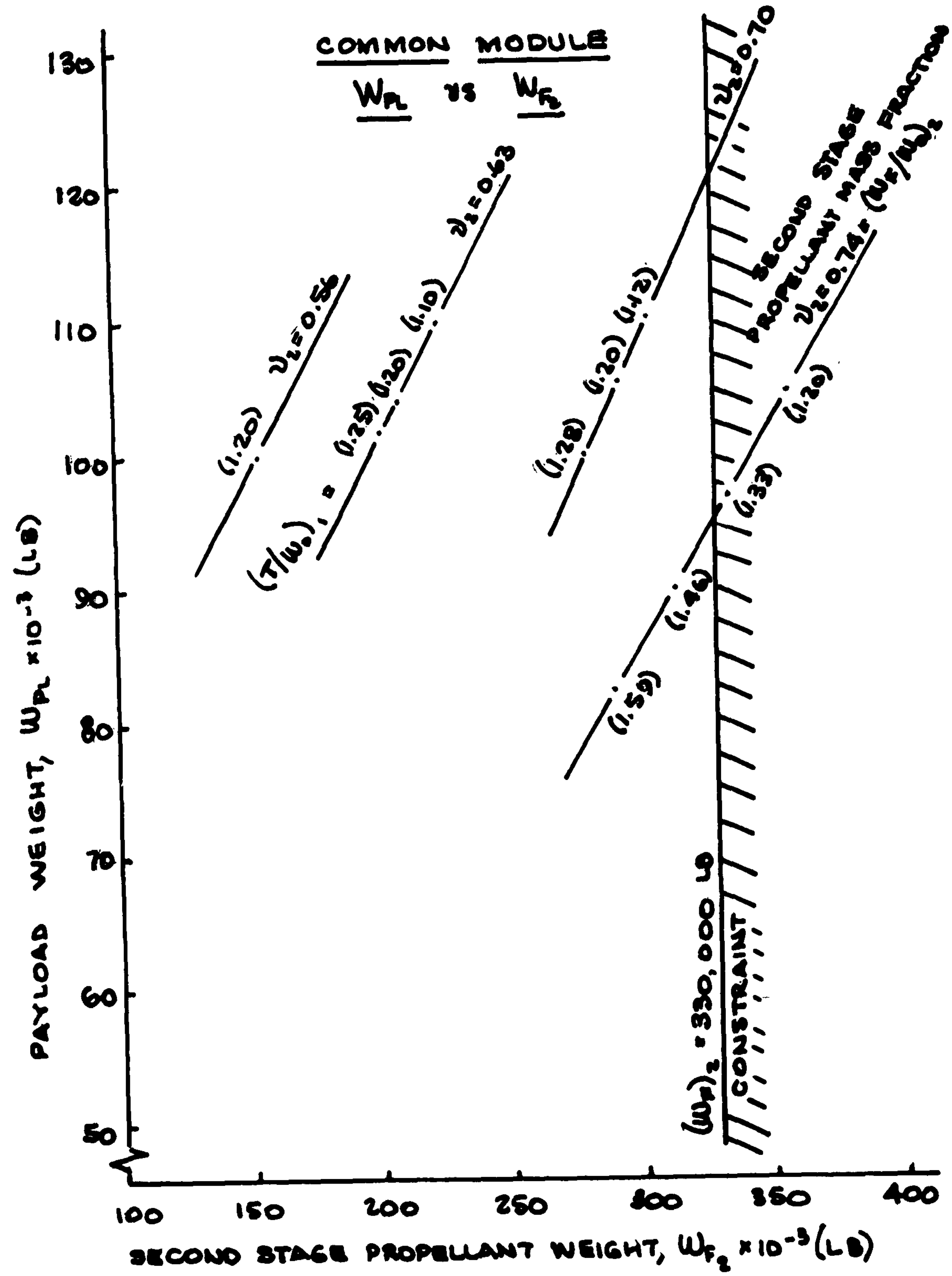


Figure 24. Payload Weight Versus Second-Stage Propellant Weight for 3[2(0. 3)WH] / 2(0. 3)WH Common Module

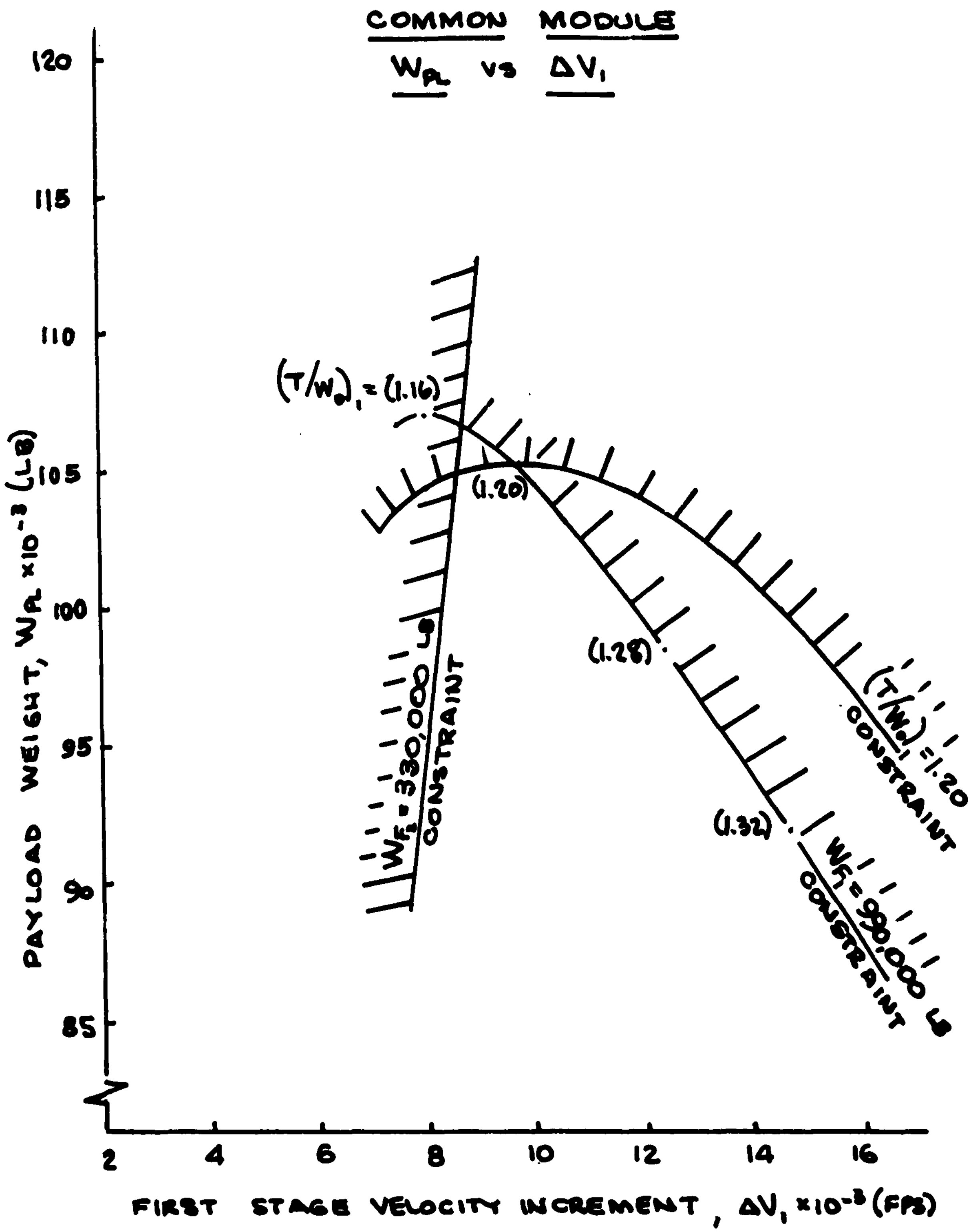


Figure 25. Payload Weight Versus First-Stage Velocity Increment for $3\left[2(0.3)WH\right]/2(0.3)WH$ Common Module

from Figure 24, for $\nu_2 = 0.70$ and $(T/W_o)_1 = 1.20$, the second-stage propellant weight is 292,000 pounds off-loaded.

For the four-common-module arrangement only one plot (Figure 26) similar to that of Figure 25 is shown. This plot is unique in that all the pertinent data can be obtained from this Figure. There is no need to go back and calculate W_{F_2} as was done in the three-common-module concept. Consequently, the four-common-module system has its maximum payload capability (139,000 pounds) at the following conditions:

1. $(T/W_o)_1 = 1.25$

2. $\Delta V_1 = 10,700$ fps

3. $W_{F_1} = 1,320,000$ pounds (capacity)

4. $W_{F_2} = 330,000$ pounds (capacity)

As was the case for the four-common-module arrangement only one plot, Figure 27, is presented for the seven-common-module system. Shown is the variation in W_{PL} and ΔV_1 for various $(T/W_o)_1$. The three appropriate system constraints have also been identified as

1. $(T/W_o)_{1_{min}} = 1.20$

2. $W_{F_1} = 2,310,000$ pounds

3. $W_{F_2} = 330,000$ pounds

The maximum payload orbited is equal to 234,000 pounds and is located at the locus of the W_{F_1} and W_{F_2} constraints. This particular seven-common-module system has identified with itself the following additional characteristics:

1. $(T/W_o)_1 = 1.35$

2. $\Delta V_1 = 14,000$ fps

3. $W_{F_1} = 2,310,000$ lb (capacity)

4. $W_{F_2} = 330,000$ lb (capacity)

It should be noted that the maximum payload capability for the three-common-module concept has been dictated by the $(T/W_o)_1$ and W_{F_1} constraints. For both the four- and seven-common-module arrangement the W_{F_1} and W_{F_2} constraints have dictated the payload capability. Actually for any common-module arrangement, the maximum payload is orbited when the

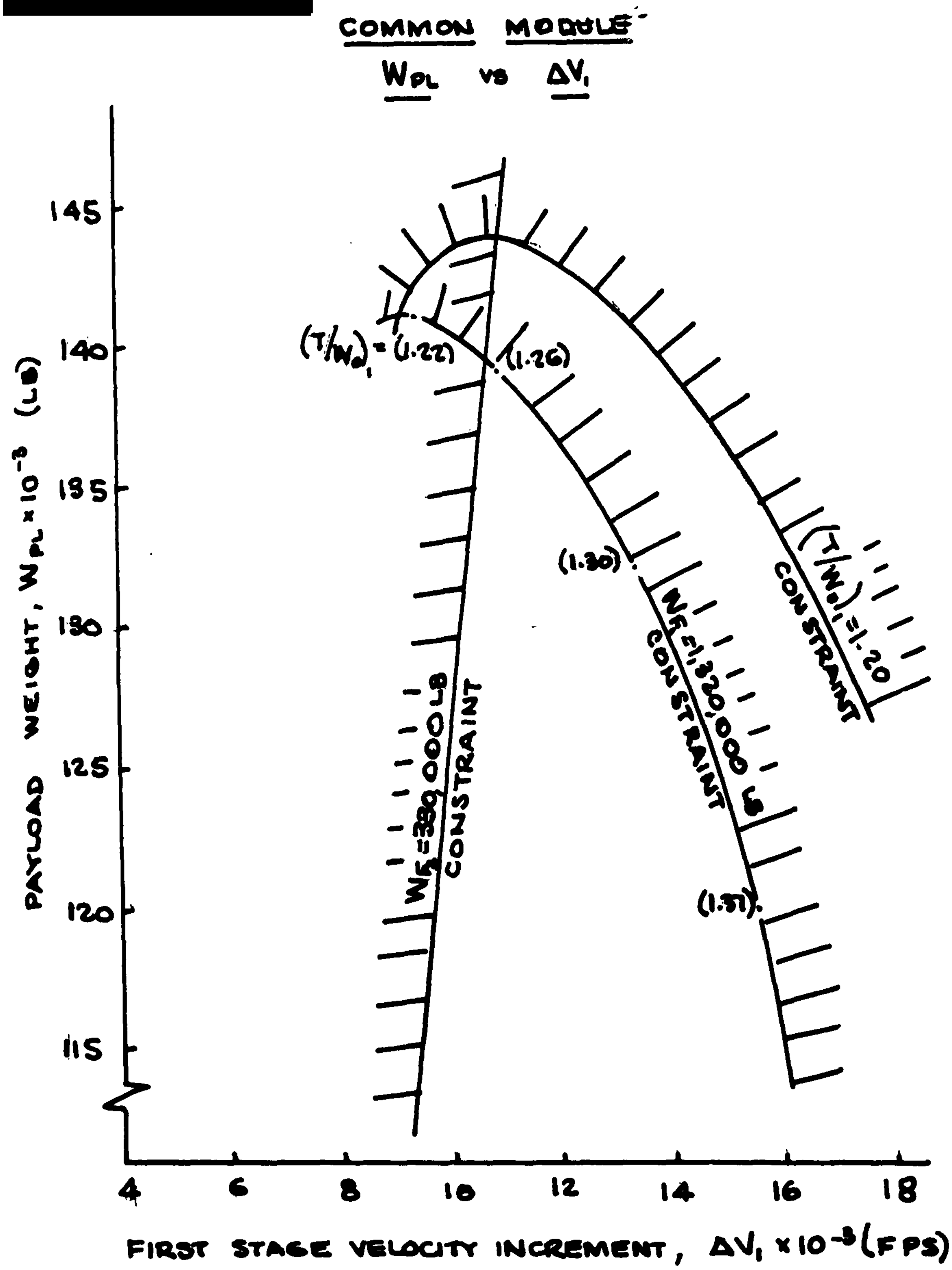


Figure 26. Payload Weight Versus First-Stage Velocity Increment for 4 [3(0. 2WH] /3(0. 2)WH Common Module

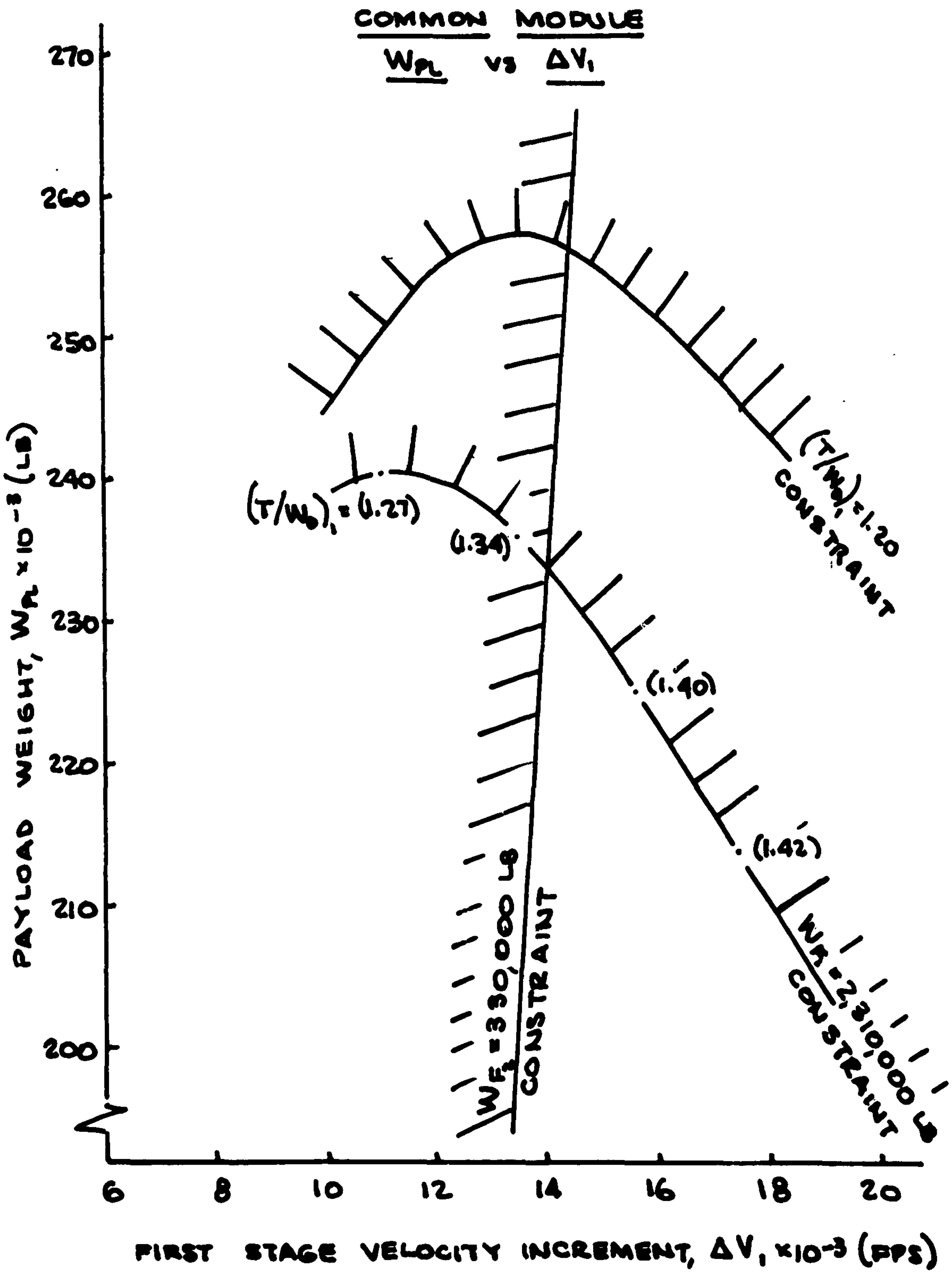


Figure 27. Payload Weight Versus First-Stage Velocity Increment for 7 [2(0. 3)] / 2(0. 3)WH Common Module

locus of the $(T/W_o)_1$, W_{F_1}, and W_{F_2} constraints is a point. This condition does not exist for any of the common-module arrangements considered, but the common three- and four-module systems do come the closest. Figure 28 presents the variation in payload weight with a change in the number of first-stage modules (N_1). Considering only the portion applicable to the common modules, it can be seen that a linear relationship exists between W_{PL} and $N_1 < 4$. When the payload capability dictated by actual flight mechanics of a seven-common-module arrangement is compared with that payload obtained by a linear extrapolation from the other common-modular concepts, the seven-common-module version loses its attractiveness. As a result of this fact coupled with its lower reliability, the seven-module concept has been omitted from further study.

Clustered First Stage Modules With Cost Optimized Second Stage

Four 600K modular systems based upon the selected 600K system have been established. These modular systems are 3 [2(0.3) WH] / 2(0.3) W'H; 3 [2(0.3) WH] / 3(0.3) W'H; 4 [2(0.3) WH] / 3(0.3) W'H; and 4 [2(0.3) WH] / 4(0.3) W'H.

Two upper-stage thrust levels have been selected for each of the two modular concepts. Since the entire system was to be cost optimized, there was no way of knowing beforehand the optimum thrust level of the second stage. Experience gained from the initial part of the Cost Optimized Booster Systems performance studies indicated the approximate second-stage thrust level, hence the above selections. The ground rules for this particular phase of the study are: (1) $(T/W_o)_1$ can assume any value so long as it does not become less than 1.20; and (2) The propellant weight per first stage module can be equal to or less than 330,000 pounds, but cannot exceed this weight. The propellant weight for the second stage will be optimized.

These ground rules are similar to those stated under the common module concept; the only difference is that the common-module constraint of W_{F_2} = 330,000 pounds has been removed.

For the 3 [2(0.3) WH] / 2(0.3) W'H configuration, Figure 29 presents W_{PL} vs W_{F_1} for various υ_2. A crossplot of the data in Figure 29 is presented in Figure 30 plotted versus ΔV_1. The orbited payload is equal to 119,900 pounds, determined from the intersection of $(T/W_o)_1$ and W_{F_1} constraints. For this payload, $(T/W_o)_1 = 1.20$, $\Delta V_1 = 9300$ fps, and W_{F_1} = 990,000 pound (capacity). From Figure 29, υ_2 is approximately 0.685 (locus of $(T/W_o)_1$ and W_{F_1} constraints), or W_{F_2} = 349,000 pounds.

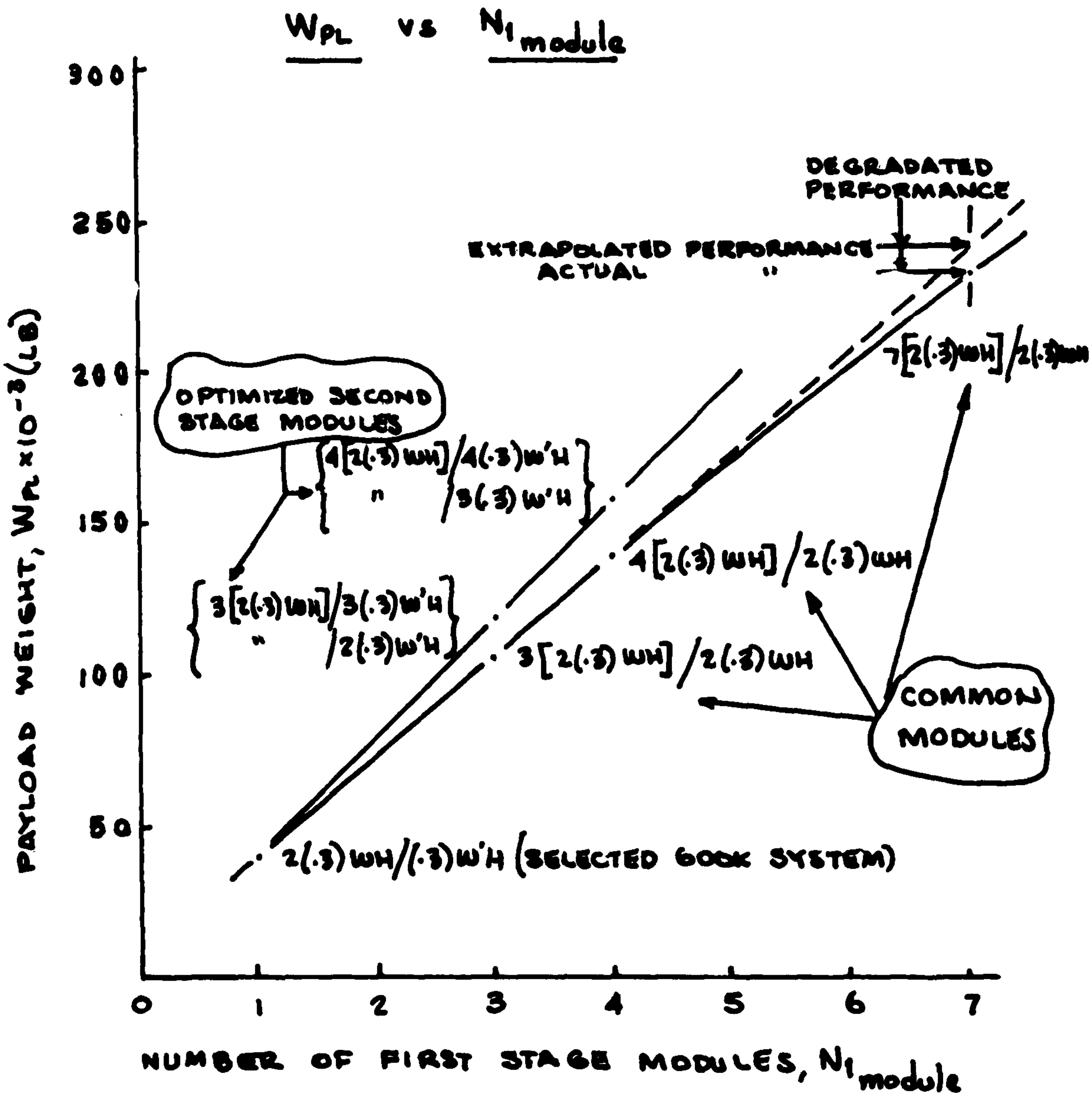


Figure 28. Payload Weight Versus Number of First-Stage Modules for Module Optimization

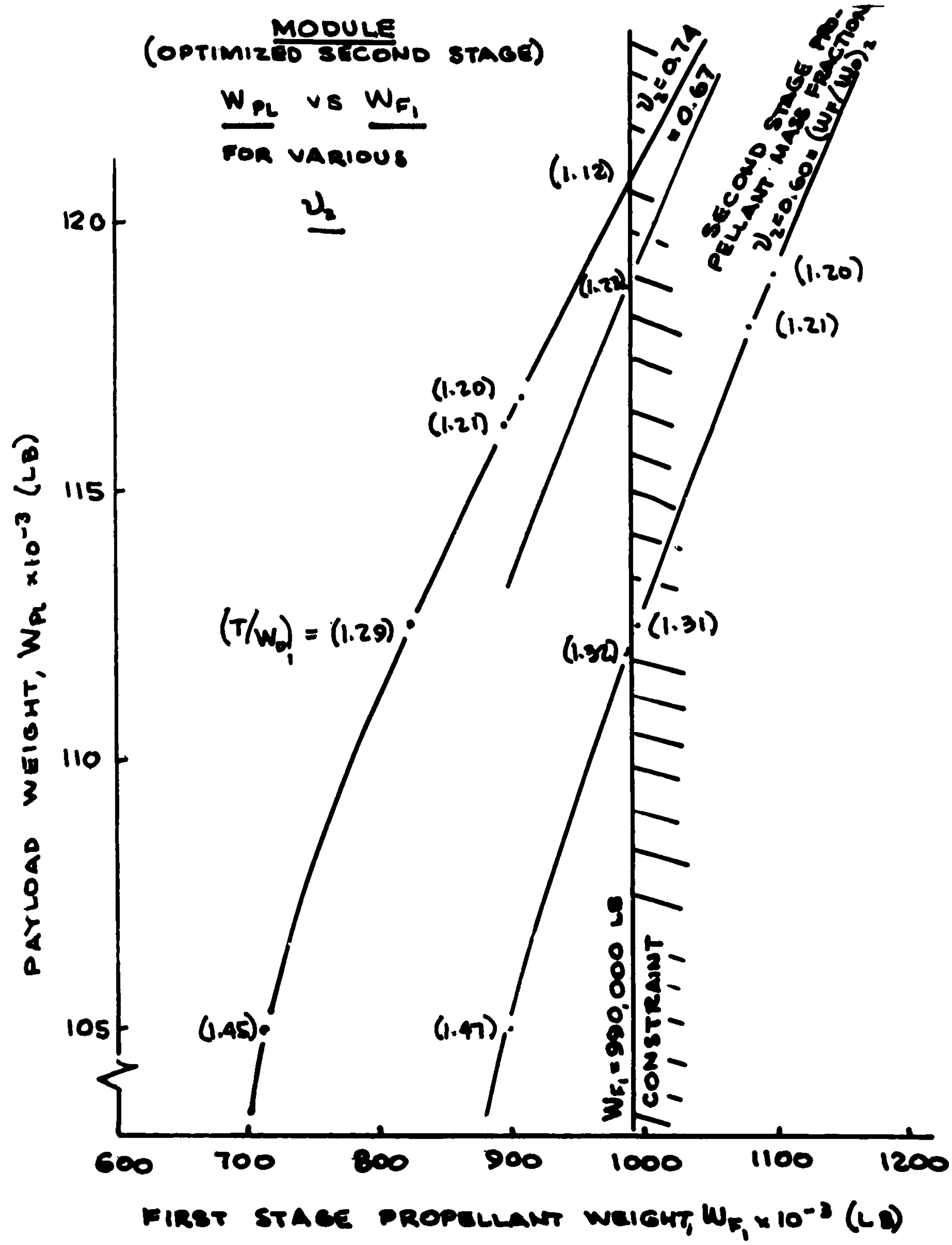


Figure 29. Payload Weight Versus First-Stage Propellant Weight for Various Second-Stage Propellant Mass Fractions for 3[2(0. 3)WH]/2 (0. 3)W'H Module (Optimized Second Stage)

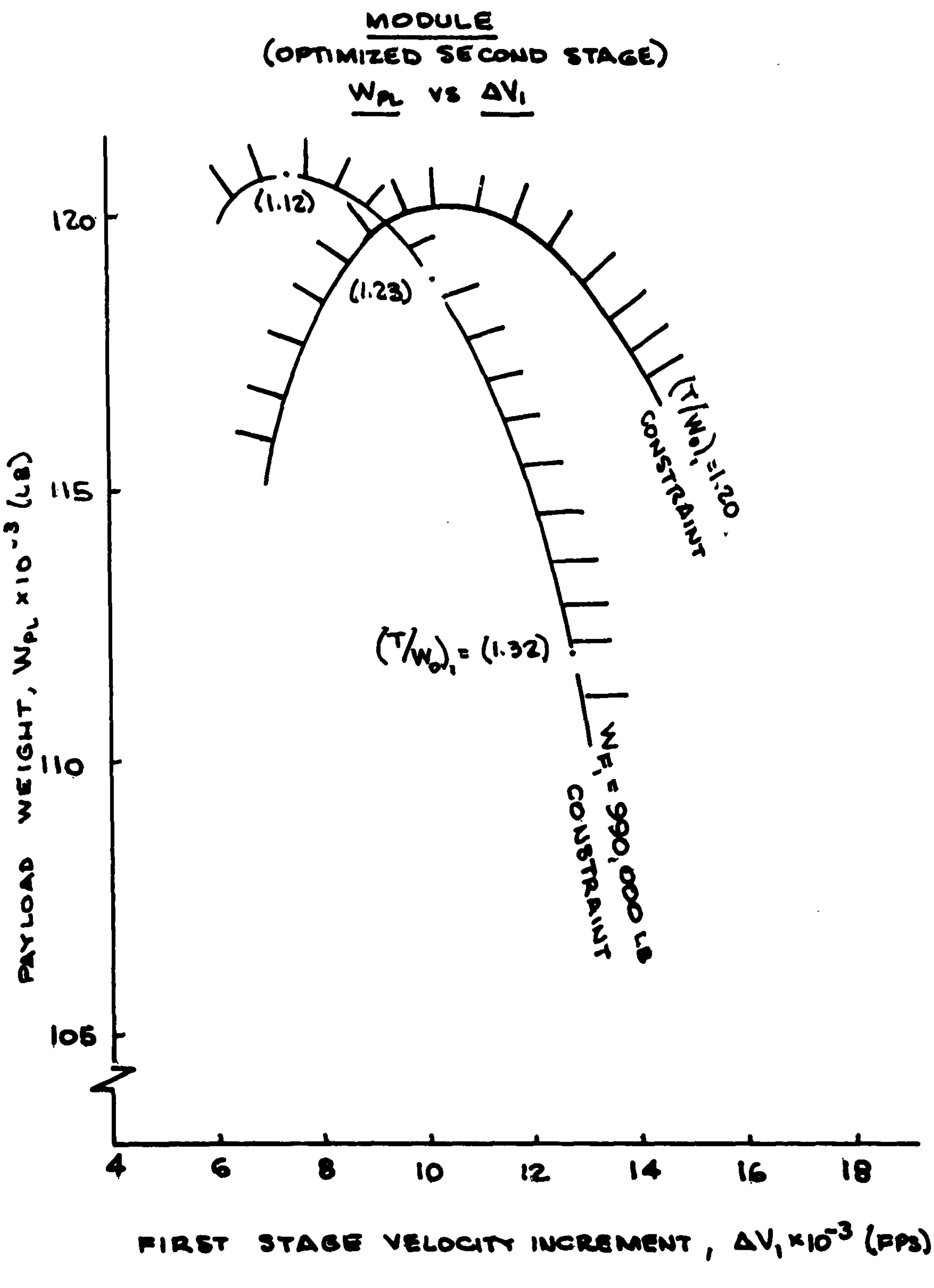


Figure 30. Payload Weight Versus First-Stage Velocity Increment for $3[2(0.3)WH]/2(0.3)W'H$ Module (Optimized Second Stage)

Much the same type of information is presented in Figures 31 and 32 where the maximum payload capability is 120,000 pounds for the 3[2(.3) WH]/ 3(0.3) W'H system. This capability is achieved when:

1. $(T/W_o)_1 = 1.20$

2. W_{F_1} = 990,000 pounds (capacity)

3. ΔV_1 = 9000 fps

4. $\upsilon_2 = 0.685$ (W_{F_2} = 349,000 pounds)

The payload for this configuration (120,000 pounds) for all practical purposes is equal to that of the previous system's 119,000 pounds. Since no payload increase was obtained from the additional second-stage thrust level of the 3[2(0.3) WH]/ 3(0.3) W'H vehicle, it may be concluded that the 3[2(0.3) WH]/ 2(0.3) W'H vehicle is the better choice. Therefore, only this latter choice will be considered in subsequent discussions.

Figures 33 and 34 indicate the payload capability for the 4[2(0.3) WH] / 3(0.3) W'H vehicle. From Figure 34, without violation of the $(T/W_o)_1$ and W_{F_1} constraints, the maximum payload is seen to be 159,900 pounds. This is achieved by the following vehicle requirements:

1. $(T/W_o)_1 = 1.20$

2. ΔV_1 = 9700 fps

3. W_{F_1} = 1,320,000 pounds (capacity)

4. $\upsilon_2 = 0.69$ (W_{F_2} = 470,000 pounds)

Figures 35 and 36 present the payload capability of the 4[2(0.3) WH] / 4(0.3) W'H version. For this case the maximum payload orbited is 160,700 pounds when:

1. $(T/W_o)_1 = 1.20$

2. ΔV_1 = 98,000 fps

3. W_{F_1} = 1,320,000 pounds

4. $\upsilon_2 = 0.69$ (W_{F_2} = 470,000 pounds)

Once again both modular systems orbit approximately the same payload weight. Because of this the 4[2(0.3) WH] / 4(0.3) W'H vehicle is omitted

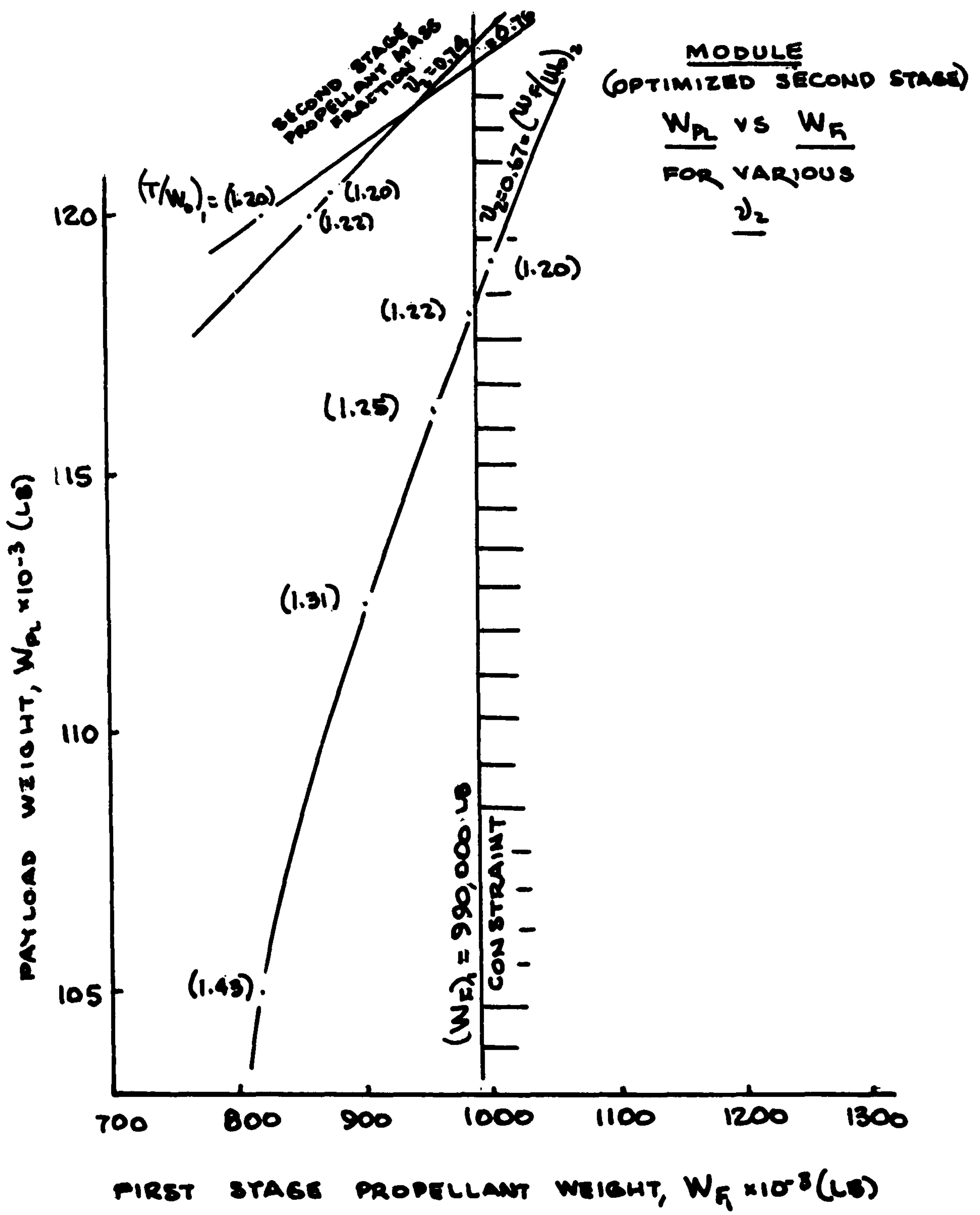


Figure 31. Payload Weight Versus First-Stage Propellant Weight for Various Second-Stage Propellant Mass Fractions for 3 [2(0. 3)WH] /3(0. 3) W'H Module (Optimized Second Stage)

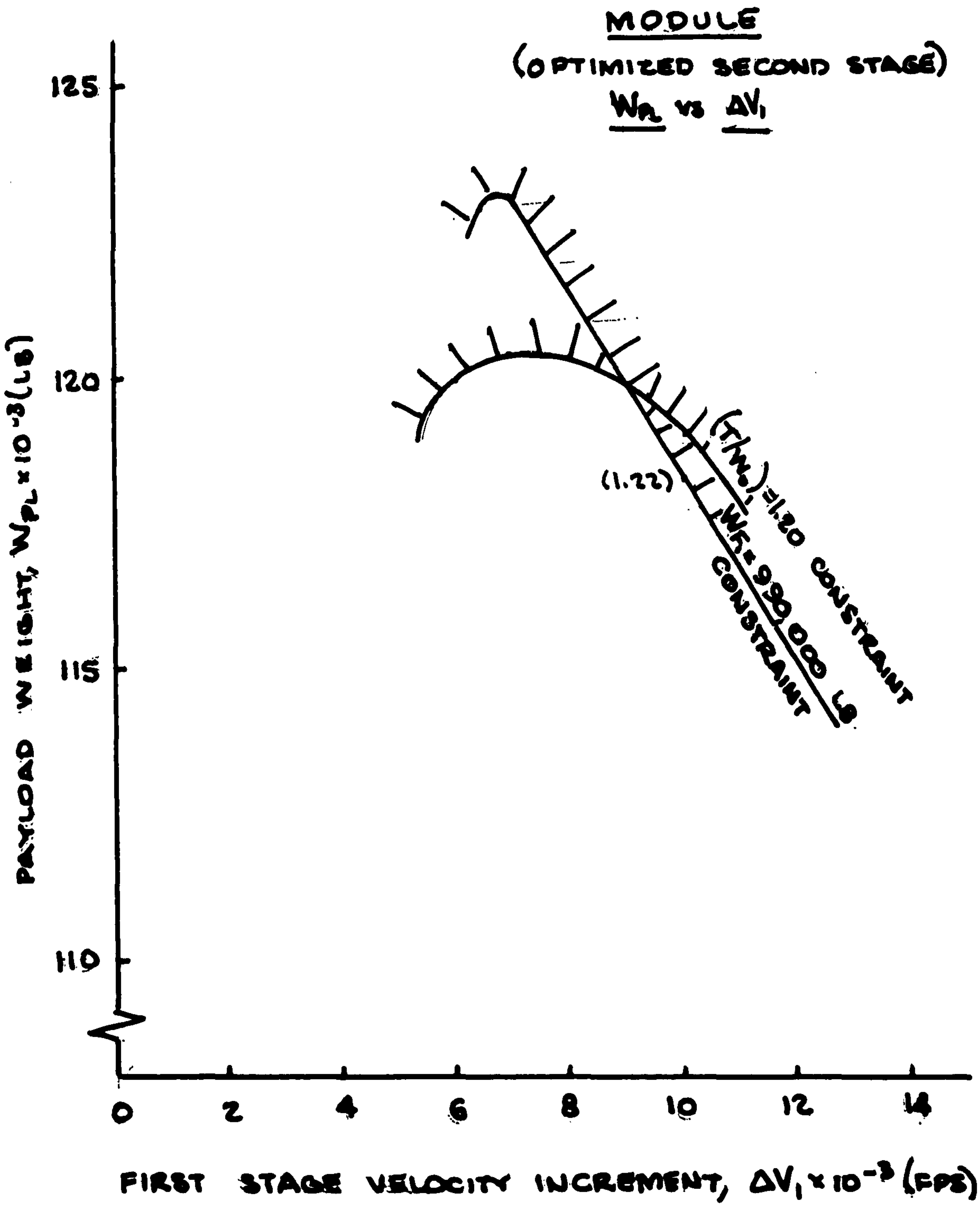


Figure 32. Payload Weight Versus First-Stage Velocity Increment for 3[2(0.3)WH]/3(0.3)W'H Module (Optimized Second Stage)

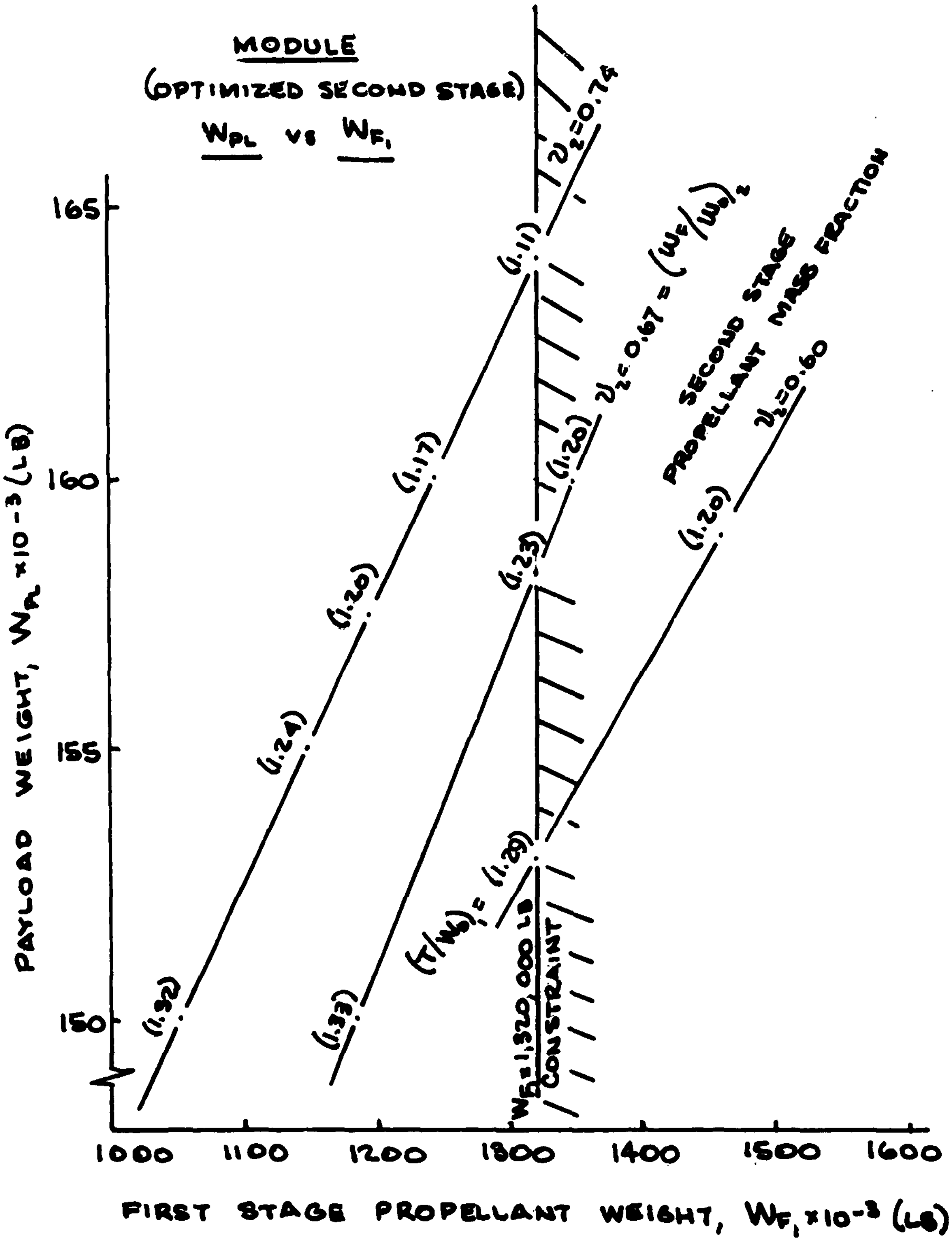


Figure 33. Payload Weight Versus First-Stage Propellant Weight for 4 [2(0.3)WH] / 3(0.3)W'H Module (Optimized Second Stage)

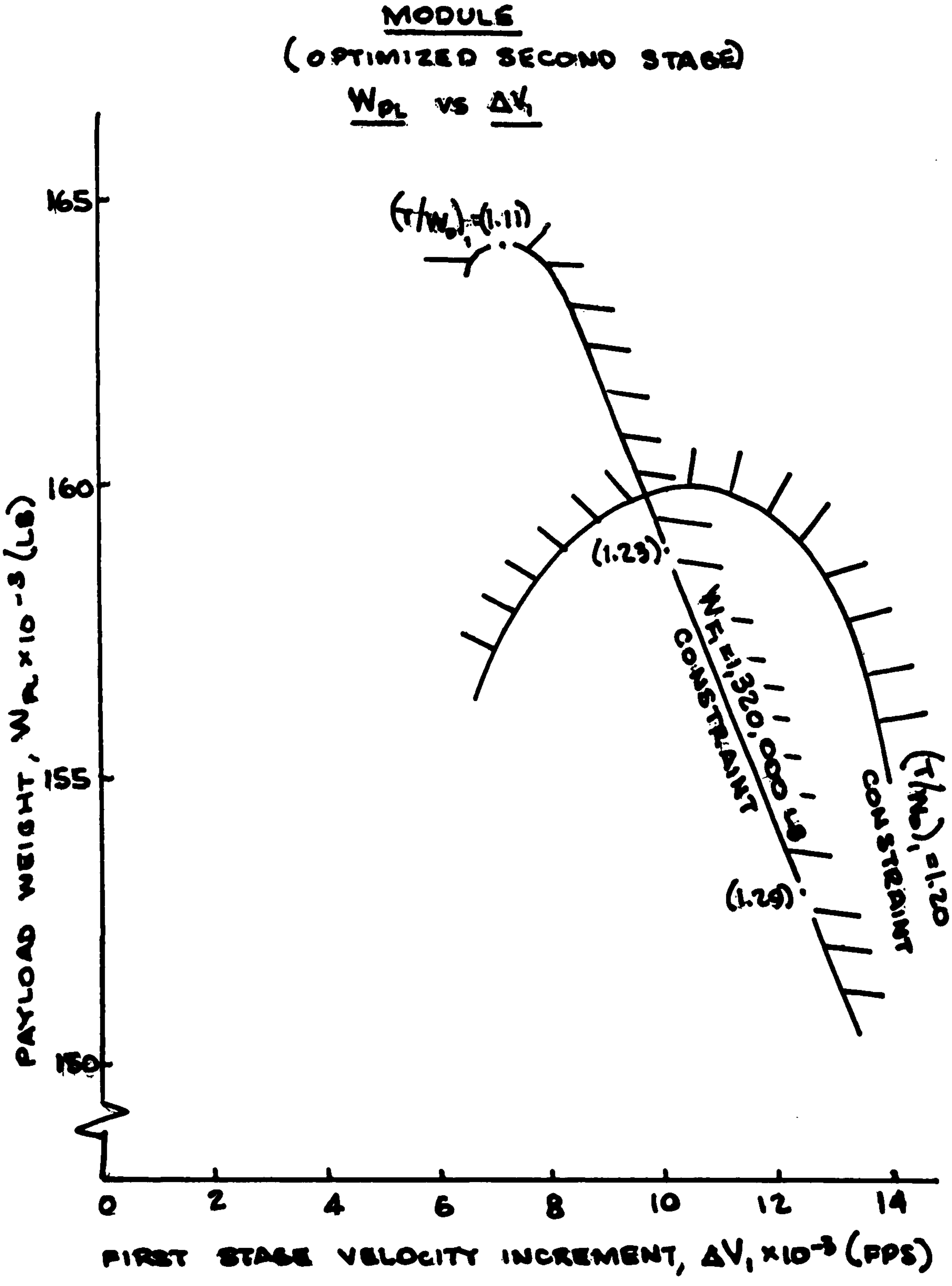


Figure 34. Payload Weight Versus First-Stage Velocity Increment for 4 [2(0. 3)WH] /3(0. 3)W'H Module (Optimized Second Stage)

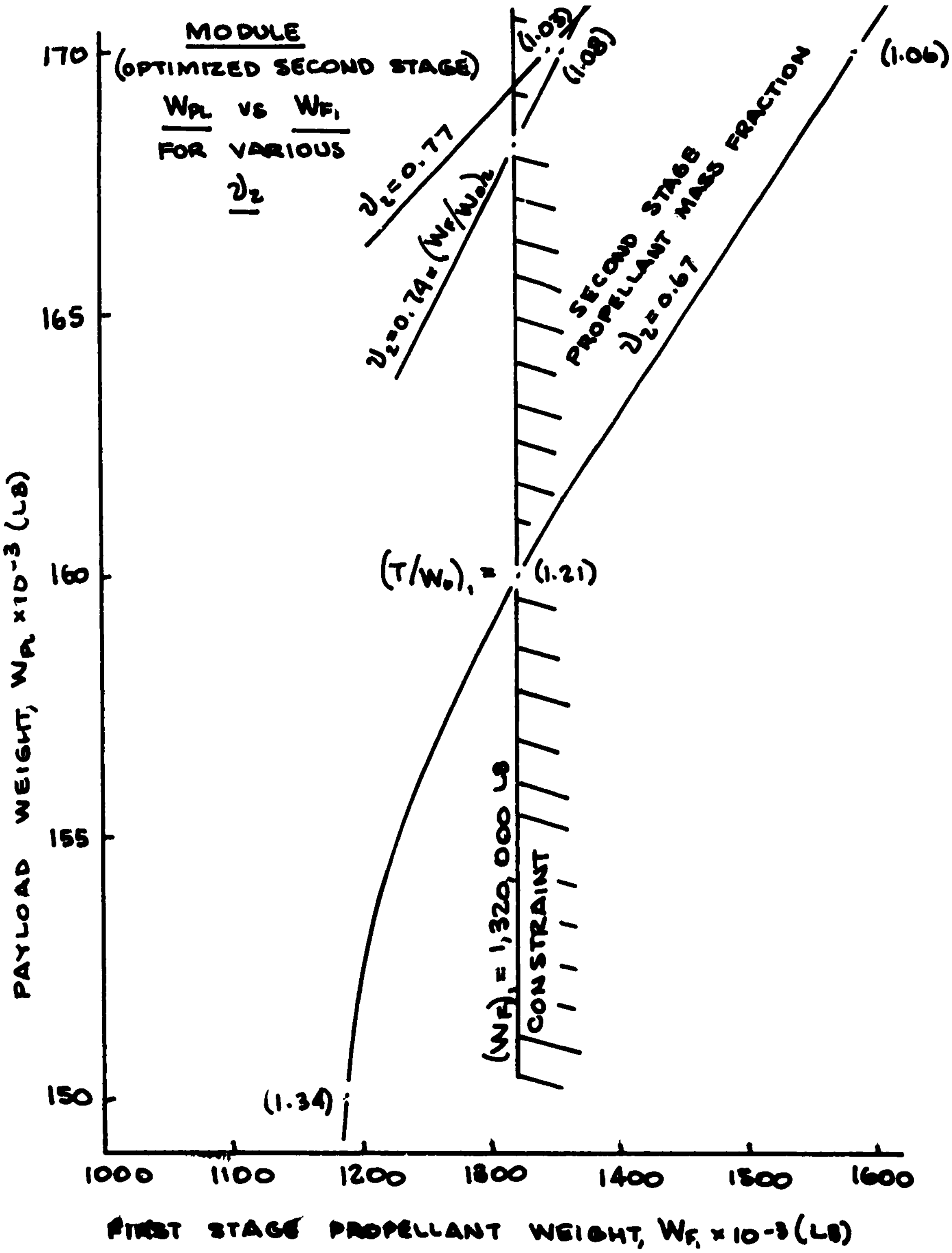


Figure 35. Payload Weight Versus First-Stage Propellant Weight for Various Second-Stage Propellant Mass Fractions for 4 [2(0.3)WH] /4 (0.3)W'H Module (Optimized Second Stage)

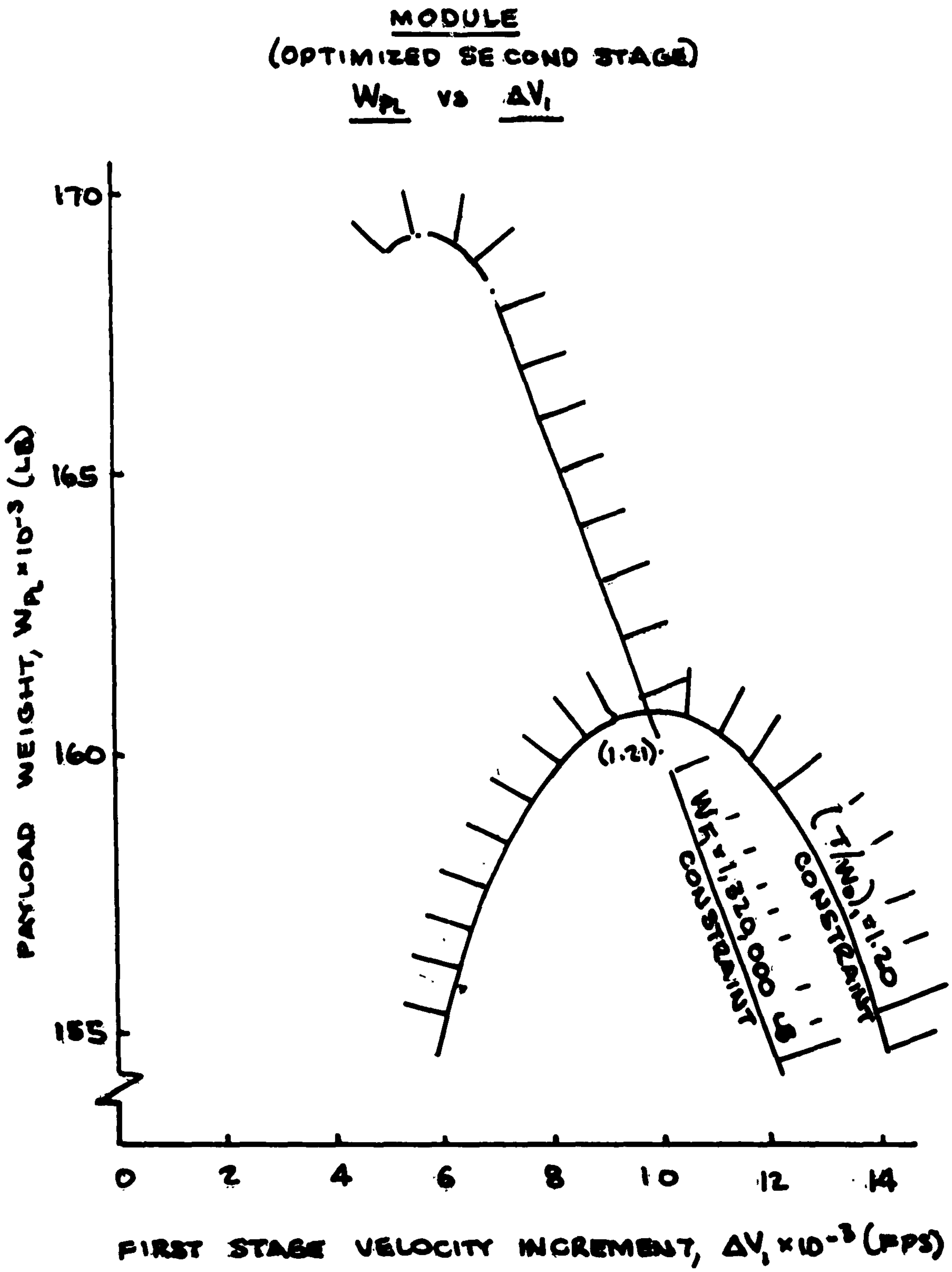


Figure 36. Payload Weight Versus First-Stage Velocity Increment for 4 [2(0. 3)WH] /4(0. 3)W'H Module (Optimized Second Stage)

from further study and only the 4 [2(0.3) WH] / 3(0.3) W'H vehicle is henceforth considered.

To complete the discussion of the modular system Figure 28 is referred to once again. The curve applicable to the common-module arrangement has already been discussed. With the curve for the performance optimized second-stage modules as shown, a comparison with a common-module set-up can be made. For a three-module arrangement, an optimized version will orbit 15,000 pounds more in payload weight than its common-module counterpart. Likewise, an optimized four-module arrangement will orbit approximately 20,000 pounds more in payload weight. Note that both curves originate from the same selected 600K system.

Segmented Solid First Stage

A configuration using a solid propellant in the first stage has been established. The object was to formulate a basis of comparison for a solid-propellant system and the 2(0.3) WH / (0.3) W'H selected system. In order not to penalize the solid-propellant system the following design philosophy was adopted: First, the solid-propellant concept would have the capability of orbiting a payload of 40,500 pounds, which would duplicate the payload capability of the 600K selected system. Second, the nozzle burning time for the solid propellant could not exceed the present technology limit of 80 seconds. Third, the initial thrust-to-weight ratio could not be less than 1.20. And finally, the second stage for both the solid-propellant and selected systems would be the same, (0.3) W'H. Following this approach the results have been presented in Figures 37 and 38. The upper portion of Figure 37 presents $(T/W_o)_1$ versus t_{b_1} for various values of υ_2. The t_{b_1} = 80 seconds constraint negates all data where the nozzle burn time is greater than 80 seconds. The lower part of Figure 37 shows the variation in W_{PL} and $(T/W_o)_1$ for various values of υ_2. Shown are the $(T/W_o)_1$ constraint and the payload weight which must be duplicated. The data of Figure 37 is valid only when the first-stage thrust level of the solid-propellant vehicle equals 1.4×10^6 pounds. Other thrust levels were analyzed with the end results shown in Figure 38, but the 1.4×10^6 pound thrust level was chosen as being the most competitive with the selected 600K all LO_2-LH_2 system. Hence, the entire results are presented only for this case and not for the other thrust levels. Figure 38 is obtained from a crossplot of Figure 37. Shown is $(T/W_o)_1$ versus υ_2 for various thrust levels. The thrust levels ranged from 1.2×10^6 pound to 1.8×10^6 pound and were analyzed in 200,000 pound increments. The data points out that the 80 second nozzle burning time constraint is violated by

$T_1 = 1.2 \times 10^6$ pound, always

$T_1 = 1.4 \times 10^6$ pound, most of the time

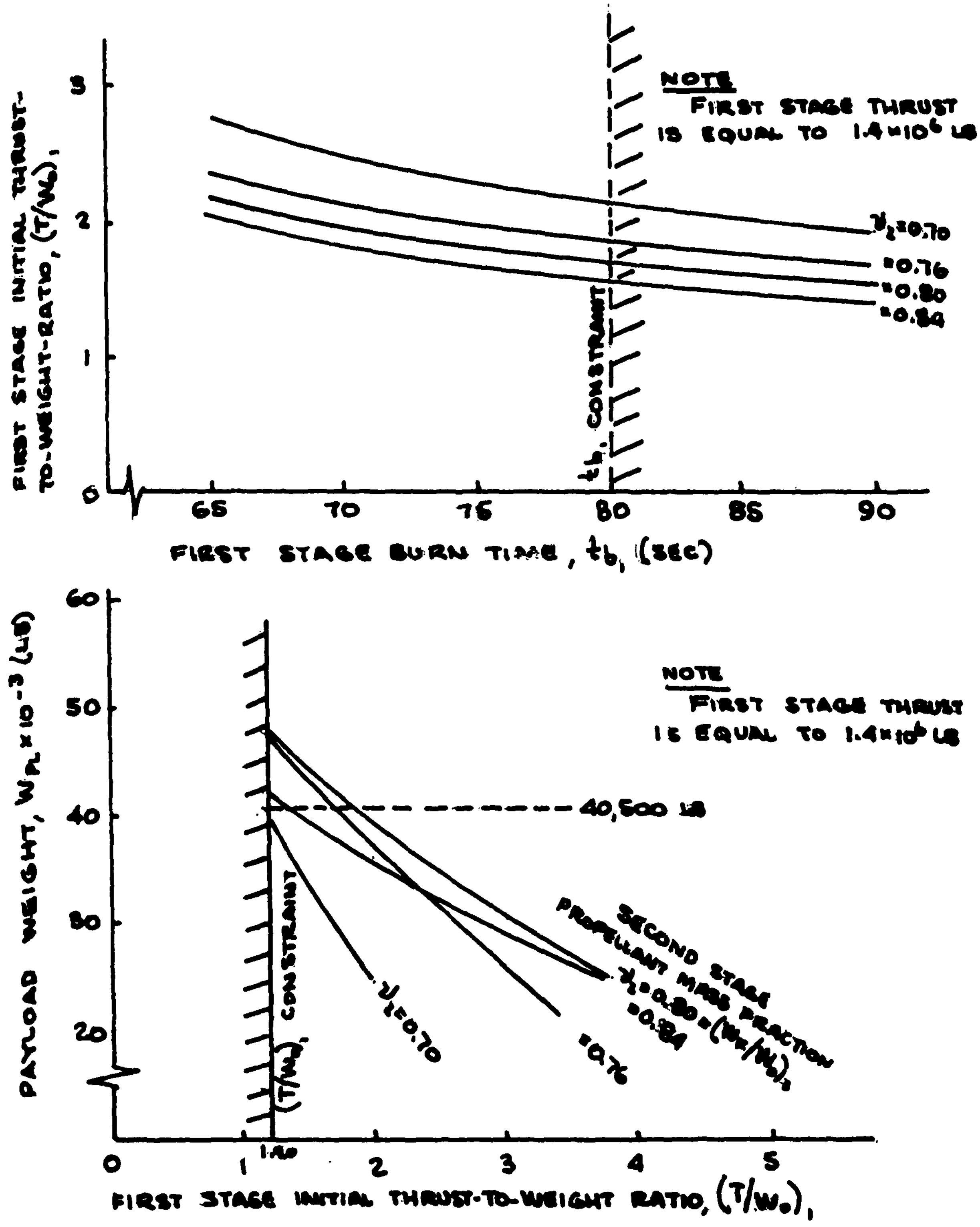


Figure 37. Payload Weight Versus First-Stage Initial Thrust-to-Weight Ratio for Solid/0.3 W'H

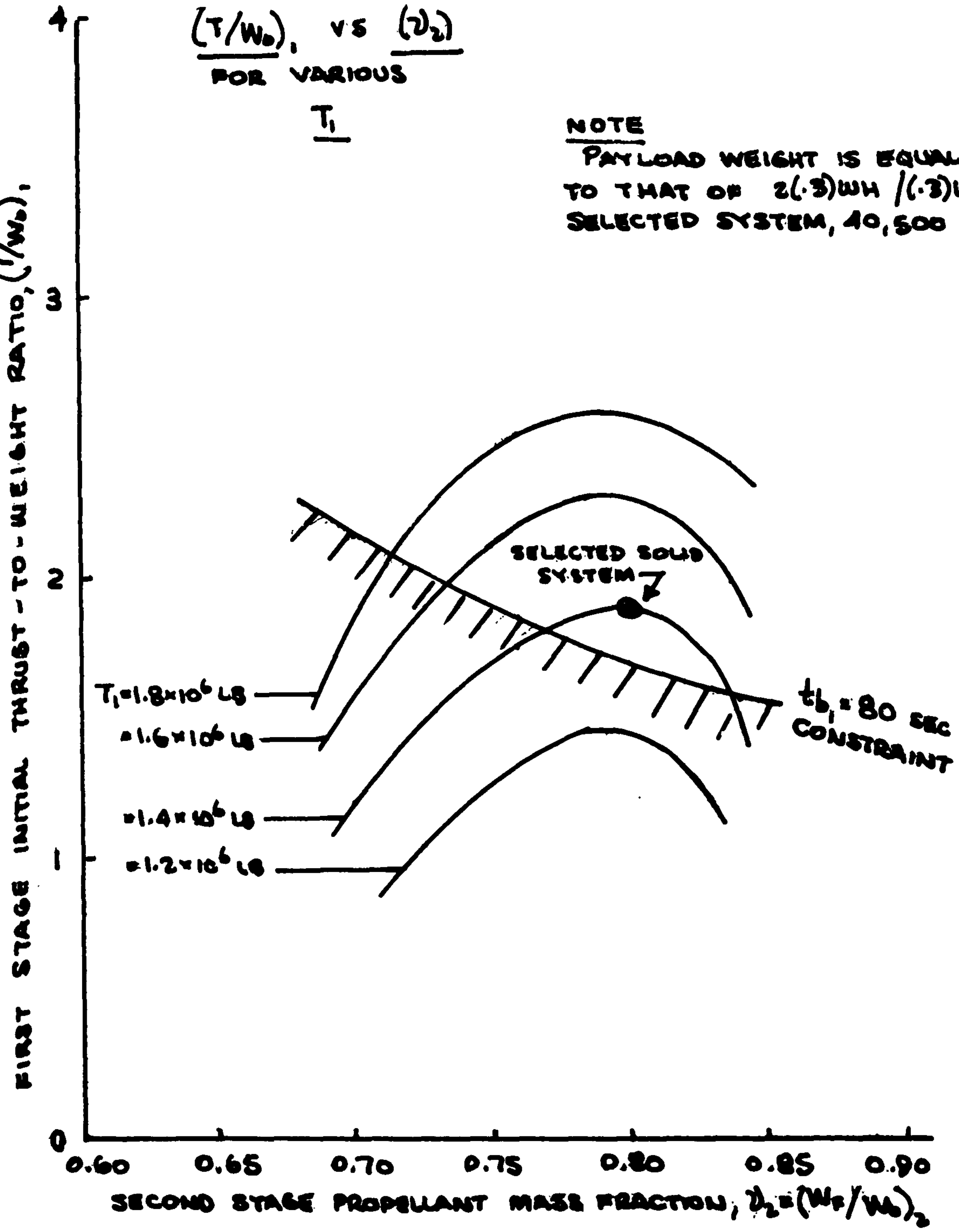


Figure 38. First-Stage Initial Thrust-to-Weight Ratio Versus Second-Stage Propellant Mass Fraction for Various First-Stage Thrusts for Solid/0.3 W'H

$T_1 = 1.6 \times 10^6$ pound, some of the time

$T_1 = 1.8 \times 10^6$ pound, practically never.

With $\nu_2 = 0.80$, the first-stage thrust level for the solid-propellant con cept was chosen to be 1.4×10^6 pound. This thrust level was felt to be the most competitive from a cost comparison evaluation with the all LO_2-LH_2 600K selected system. The solid-propellant vehicle properties are:

1. $(T/W_o)_1 = 1.90$
2. $T_1 = 1.4 \times 10^6$ pounds
3. $t_{b1} = 80$ seconds
4. $V_1 = 4900$ fps
5. $\upsilon_1 = 0.59$
6. $\upsilon_2 = 0.80$

600K Selected System-Expansion Ratio Variation

A performance comparison has been made between the selected 600K system, 2(0.3) WH/(0.3) W'H and a 600K system with identical nozzle expansion ratios in both upper and lower stages, 2(0.3) W'H/(0.3) W''H. The selected system used a first-stage expansion ratio (ϵ_1) of 23 and a second-stage expansion ratio (ϵ_2) of 90; the latter system used $\epsilon_1 = 45 = \epsilon_2$. The consideration of equal stage expansion ratios seemed reasonable in the light of further cost reductions when usage is made of identical parts. Figure 39 presents W_{PL} versus ΔV_1 for these two systems. The equal expansion ratio feature is seen to shift the performance curve to the right. Since the maximum payload capability is identical for the two systems, it can be inferred that the $\epsilon_1 = 45 = \epsilon_2$ combination is just as good a choice as the $\epsilon_1 = 23$ and $\epsilon_2 = 90$ combination. The shift to the right is explained when it is recalled that an increase in ϵ results in an increase in I_{sp} which in turn results in a velocity increase. It would appear then, that the above equal nozzle expansion ratio combination is justifiable from a standpoint of dollars-per-pound orbited.

600K Selected System-Chamber Pressure Variation

The effect of a variation in chamber pressure upon the performance capability of the selected 600K system, 2(0.3) WH / (0.3) W'H, has been calculated. In all previous analyses, the chamber pressure of the advanced P & W engine (designated WH and/or W'H) was set at 3000 psia and the

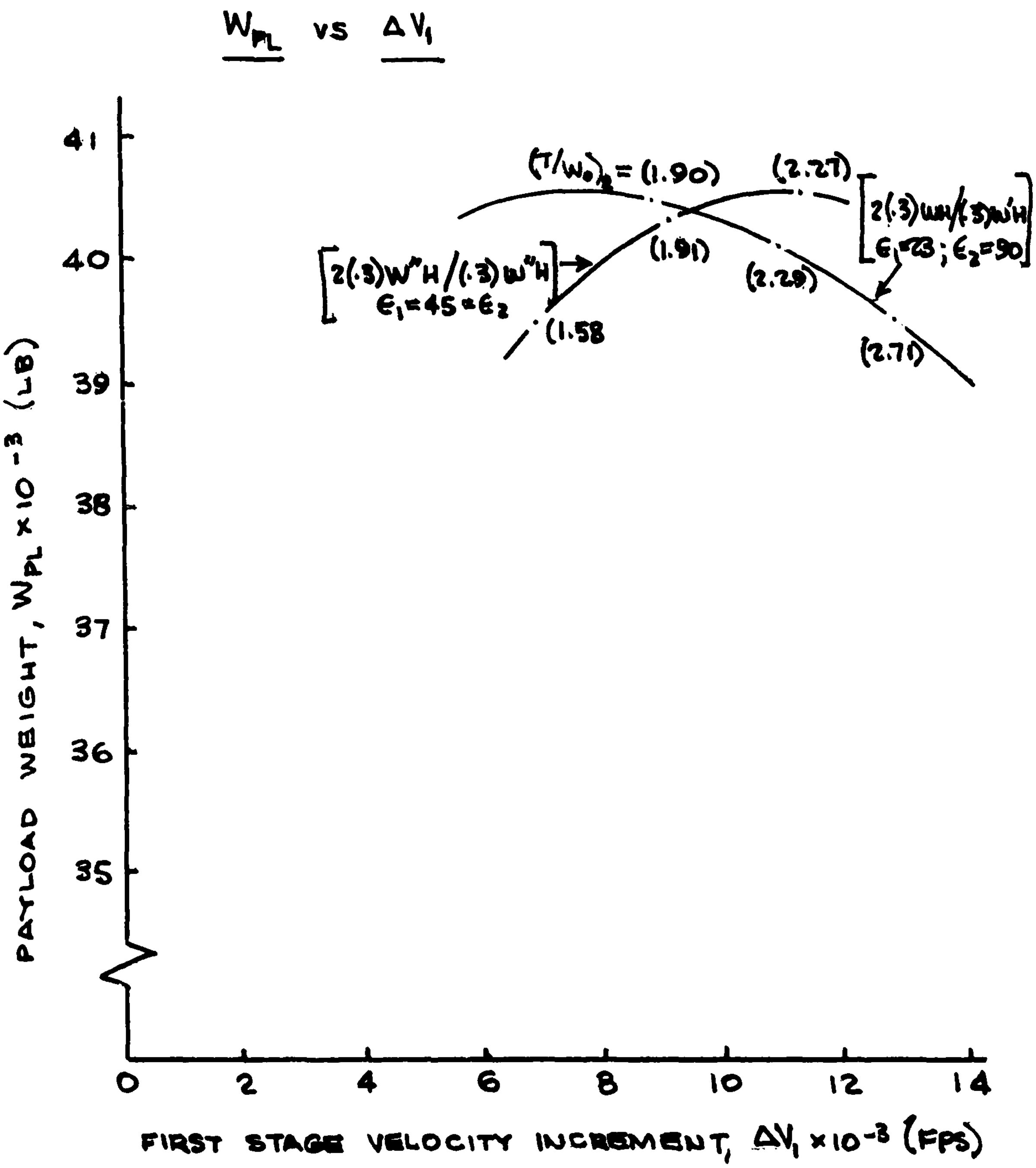


Figure 39. Payload Weight Versus First-Stage Velocity Increment for 2(0. 3)/0. 3 - Effect of Expansion Ratio

engine properties were then determined. To justify 3000 psia as an optimum selection for the high-pressure engine cycle, a chamber pressure of 5000 psia was selected to be studied and its payload capability determined. The results are shown in Figure 40 where it can be seen that the 3000 psia system always put into orbit 1000 pounds more of payload weight. Therefore, our selection of the 3000 psia engine as optimum for the high-pressure engine cycle is justified. The 5000 psia system is henceforth omitted from further study.

Liquid Fluorine Second Stages

Three vehicles with liquid fluorine (LH_2-LF_2 system) as the oxidizer of the second stage have been established. These vehicles have been compared with the 600K selected system, 2(0.3) WH / (0.3) W'H, and have been developed according to the following design criteria: $(T/W_o)_1 = 1.20$; second-stage thrust levels of 100K, 200K, and 300K would be considered; and the first stage would be a 2(0.3) WH system.

Figure 41 presents the performance capabilities of these three LF_2 systems and that of the selected 600K system. Plotted are curves of W_{PL} versus ΔV_1. The second-stage thrust level for the LF_2 system optimizes at 200,000 pounds, since it has the maximum payload capability (40,000 pounds) for any of the three fluorine vehicles. Even with this 40,000 pound capability, the optimum LF_2 falls 500 pounds short of the 40,500-pound orbited payload of the selected 600K system.

Single-Stage-To-Orbit

Two single-stage-to-orbit systems representing two different methods of approach have been established. The first of these is a psuedo single-stage-to-orbit system wherein the kick into orbit at apogee is supplied by a separate kick stage using solid propellants. The other represents a true single-stage-to-orbit where, after the long coast period, the first-stage propulsion system is relit and thereby supplies the kick into orbit at apogee.

The design philosophy adhered to in the analysis of these two systems is as follows: $(T/W_o)_1 = 1.20$. The initial thrust-to-weight ratio would equal 1.20. Each system would employ a single engine with a 600K thrust level in the first stage with LO_2-LH_2 as the first-stage propellant. Because single-stage-to-orbit vehicles are of necessity large in diameter, each system would employ secondary expansion in the first stage. This would result in achieving an overall first-stage nozzle expansion ratio of 400. Because of the difficult problem of the selection of an adequate control system for the single-stage-to-orbit vehicles considered (see Attitude Control section of this report), the performance evaluation of these vehicles would be in the form of a general parametric study. This would entail a

W_{PL} vs ΔV_1

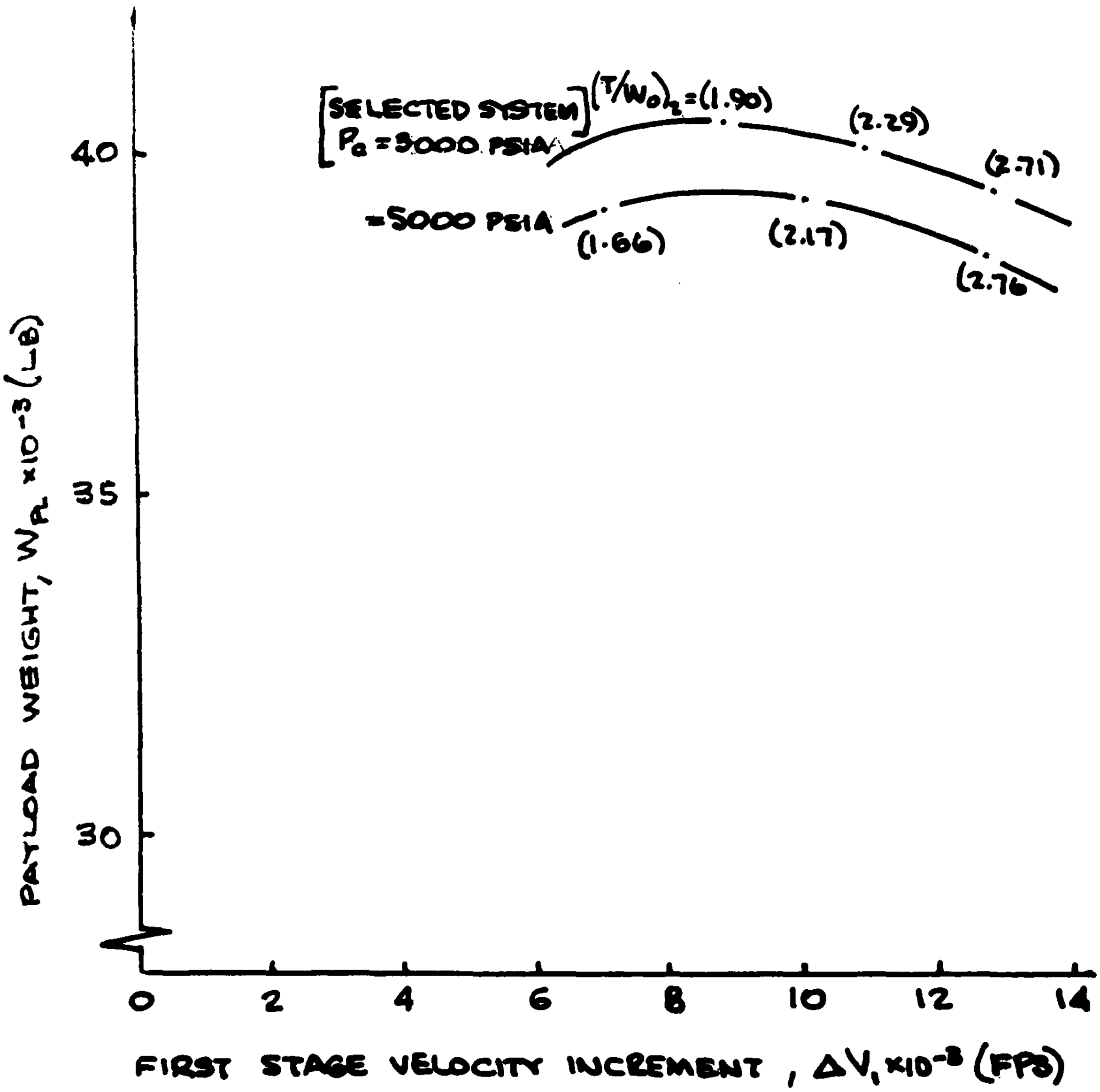


Figure 40. Payload Weight Versus First-Stage Velocity Increment for 2(0.3)WH/0.3 W'H - Effect of Chamber Pressure

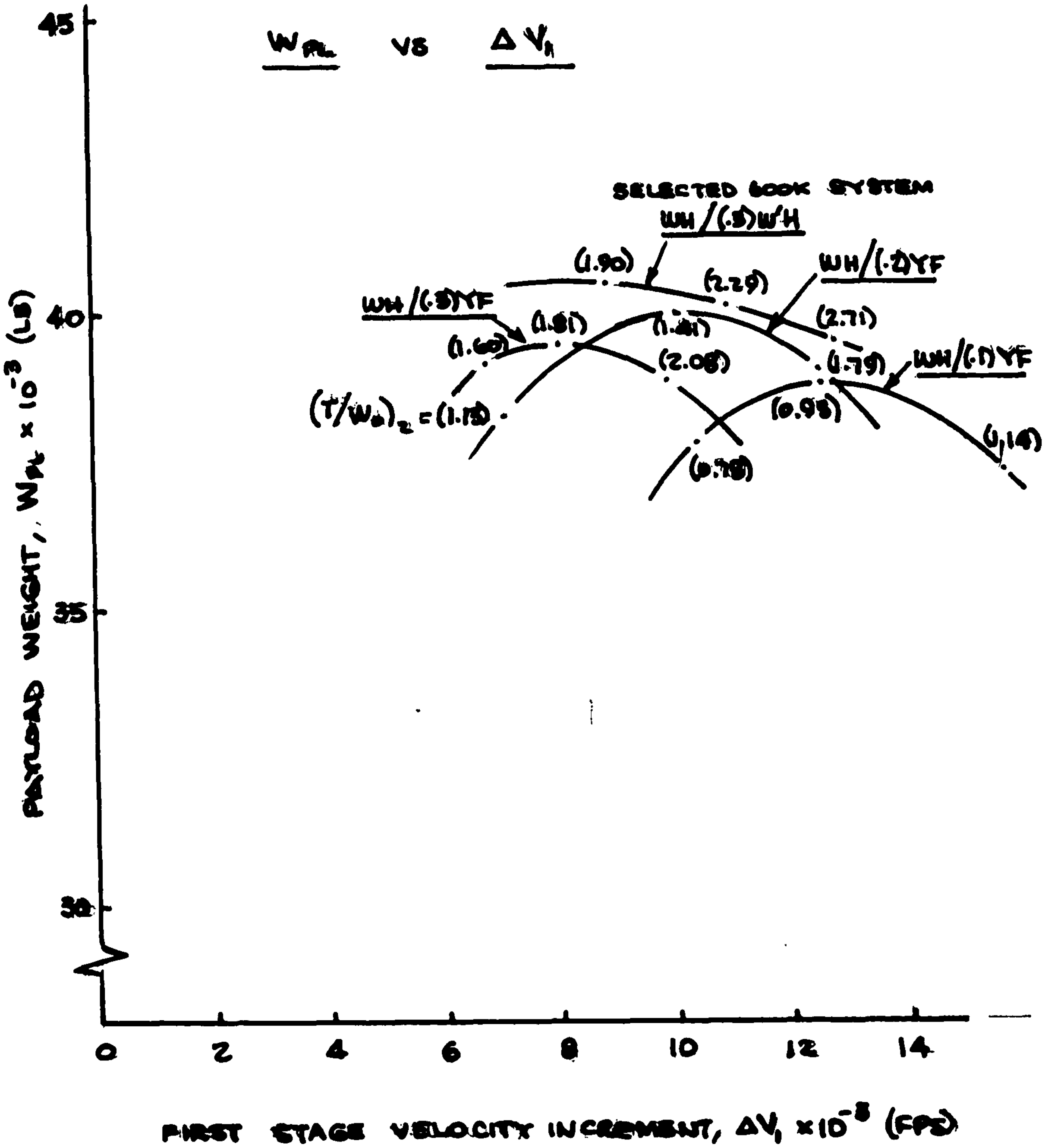


Figure 41. Payload Weight Versus First-Stage Velocity Increment for 2(0. 3)WH/Flourine

simultaneous analysis of the effect of a variation in first-stage structure mass fraction, ν_{b_1}, in addition to the usual performance evaluation as outlined in the Methodology Section. Values of ν_{b_1} ranging from 0.910 to approximately 0.945 would be considered. The results of the general performance parametric study are shown in Figure 42. The variation of W_{PL} and $(T/W_o)_1$ for several values of ν_{b_1} are presented for both the separate solid kick-stage system and the relightable first-stage system. This W_{PL} and $(T/W_o)_1$ variation is seen to be linear in shape and is a result of the constant value of the first-stage propellant mass fraction ($\nu_1 = 0.872$ or $WF_1 = 435{,}000$ pound). Such a relationship is dictated by the mechanics of the situation. First of all, ν_1 is invariant since it is dependent only on engine characteristics ($I_{sp\ avg}$) and mission profile (ΔV_{req}). From this point on, W_{PL} is a direct function only of ν_{b_1} and initial take-off weight, W_{o1}; hence, the justification of the linear trade-off between W_{PL} and $(T/W_o)_1$ for various values of ν_{b_1} as shown in Figure 42. Figure 43 is obtained from a crossplot of the data shown in Figure 42. Presented is a plot of W_{PL} versus ν_{b_1} for both single-stage-to-orbit systems when $(T/W_o)_1 = 1.20$. It is apparent from this plot that the concept employing the separate solid kick-stage puts into orbit approximately 5500 pounds more payload weight than the relightable first-stage version. Had not pressurization, chill-down, and evaporation losses of the latter system amounted to 5000 pounds, that system would have been much more competitive with the separate solid kick-stage system where no such losses exist.

Paraglider Recovery

The effect of recovery gear necessary for first-stage booster recovery by means of a paraglider has been investigated for the following four configurations:

1. 2(0.3) WH - (PR) / (0.3) W'H
2. (0.6) WH - (PR) / (0.1) W'H
3. (0.6) WH - (PR) / (0.05) W'H
4. 2(0.75) WH - (PR) / (0.75) W'H

where (PR) is defined as paraglider recovery. The first and last configurations are termed the conventionally staged recovery systems. The only difference between these two systems and the 600K and 1500K selected systems is the addition of the paraglider gear. The vehicle characteristics are identical to those determined in the staging optimization studies for the 600K and 1500K selected systems. The second and third configurations are termed the extended first-stage recovery systems. For these two

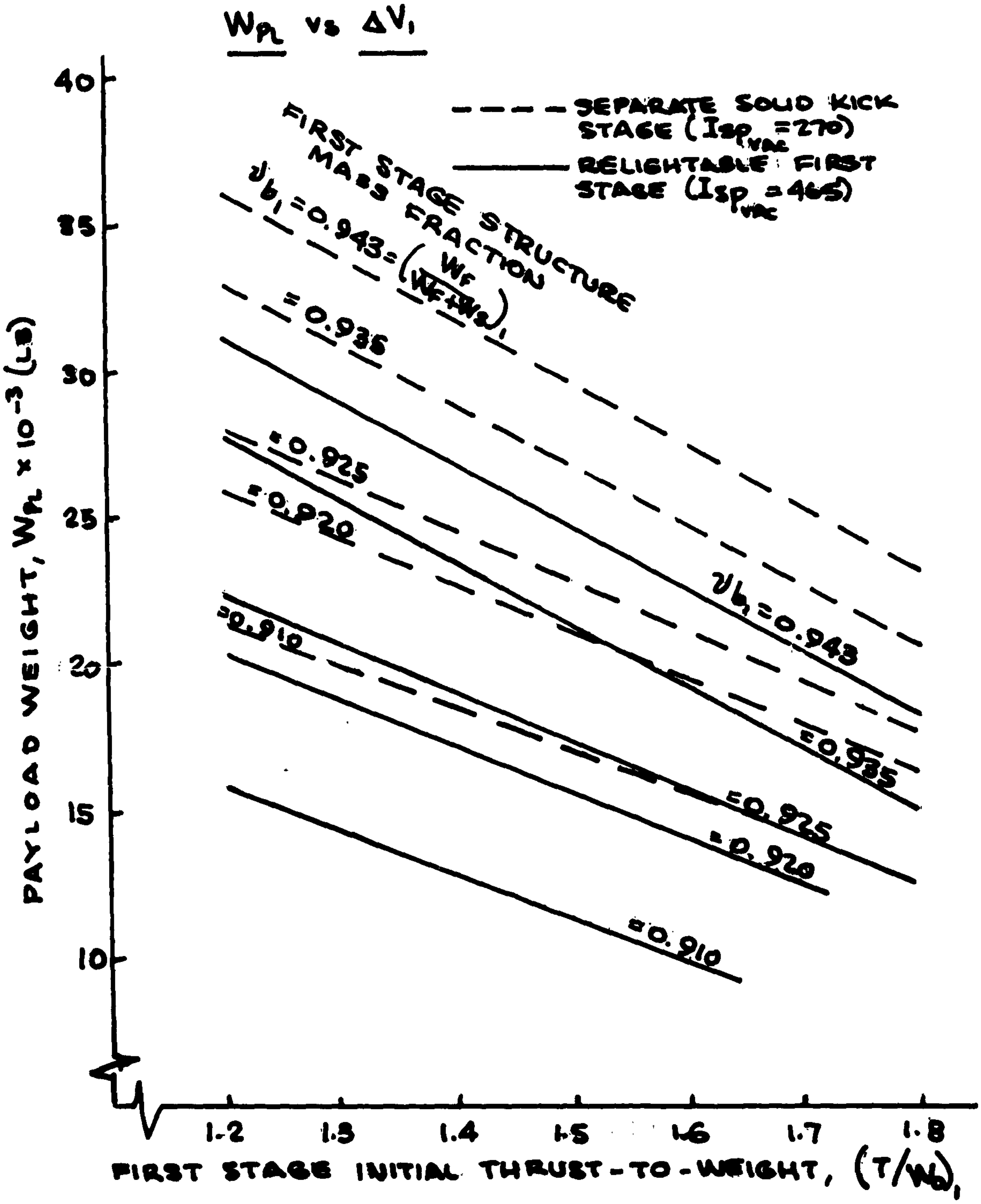


Figure 42. Payload Weight Versus First-Stage Initial Thrust-to-Weight for 0.6 W'''H - Single-Stage-to-Orbit

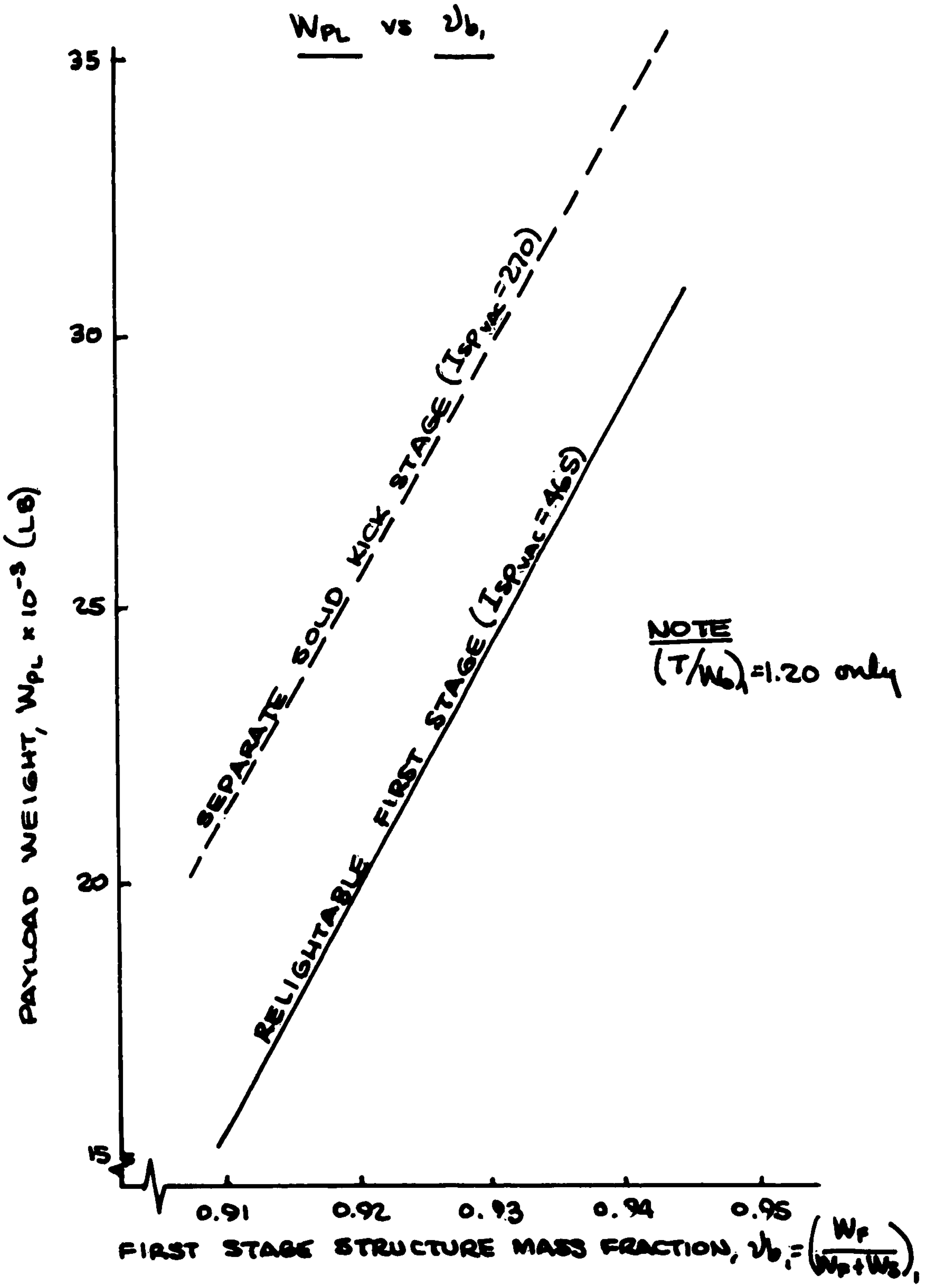


Figure 43. Payload Weight Versus First-Stage Structure Mass Fraction for 0.6 W'''H - Single-Stage-to-Orbit

configurations staging is extended past the point determined to be optimum in the initial staging optimization studies. The philosophy is to recover a larger first stage and thereby reduce the system cost.

Figure 44 presents the payload capabilities for the three 600K paraglider recovery configurations. In comparison to the 600K selected system, the 600K conventionally staged recovery system suffers a payload degradation of 1000 pounds. Because of the large cost reductions foreseen and only the small payload degradation, the conventionally staged recovery scheme appears very attractive. Also presented in Figure 44 is the effect on payload of extending the staging point of the first stage and its subsequent recovery. It can be seen that the further the staging is extended, the more the payload is degraded, until in the limit the recovery of a single-stage-to-orbit vehicle is approached. The method of analysis for the results of the above was to alter the first-stage structure mass fraction, υ_{b_1}, according to the following relationship:

$$\upsilon^*_{b_1} = \frac{\upsilon_{b_1}}{1 + 0.16\,(-\upsilon_{b_1})}$$

where $\upsilon^*_{b_1}$ refers to the booster-paraglider combination. The equation states that the total weight of recovery gear is equal to 16 percent of the end-boost weight of the booster without the paraglider, i.e. $W^*_{EB_1} = 1.16/W_{EB_1}$.

Figure 45 presents the effect on payload when a 1500K conventionally staged recovery system is employed. Plotted is the variation of W_{PL} with ΔV_1. The data indicates a 2600-pound reduction in orbited payload weight by incorporation of the recovery gear. This recovery system still appears attractive in light of the possible cost reductions. The method of analysis was identical to that used in the 600K recovery system studies. No extended first-stage recovery data was generated for the 1500K systems.

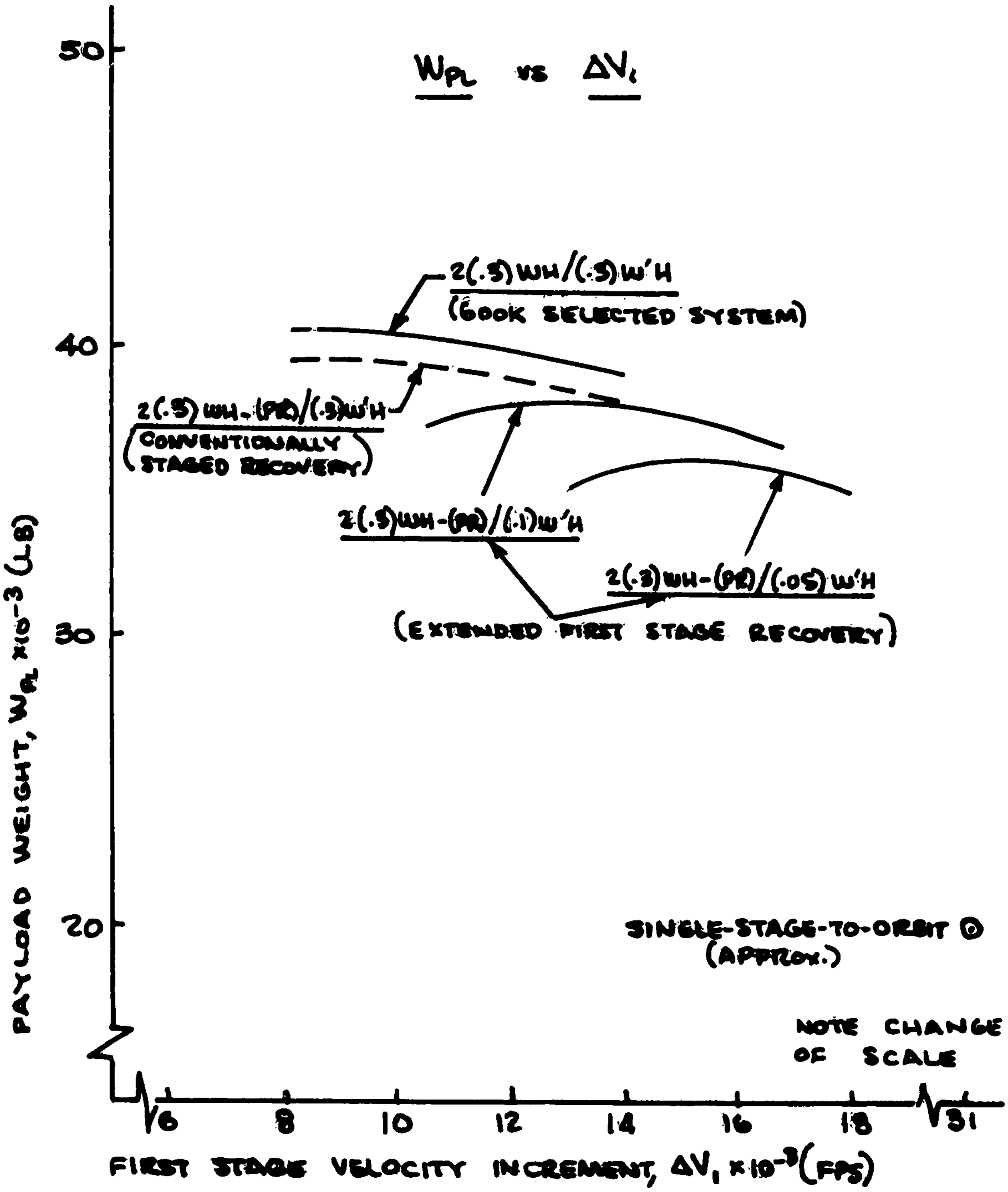


Figure 44. Payload Weight Versus First-Stage Velocity Increment for 2(0.3)/0.3 W'H - Effect of Paraglider Recovery

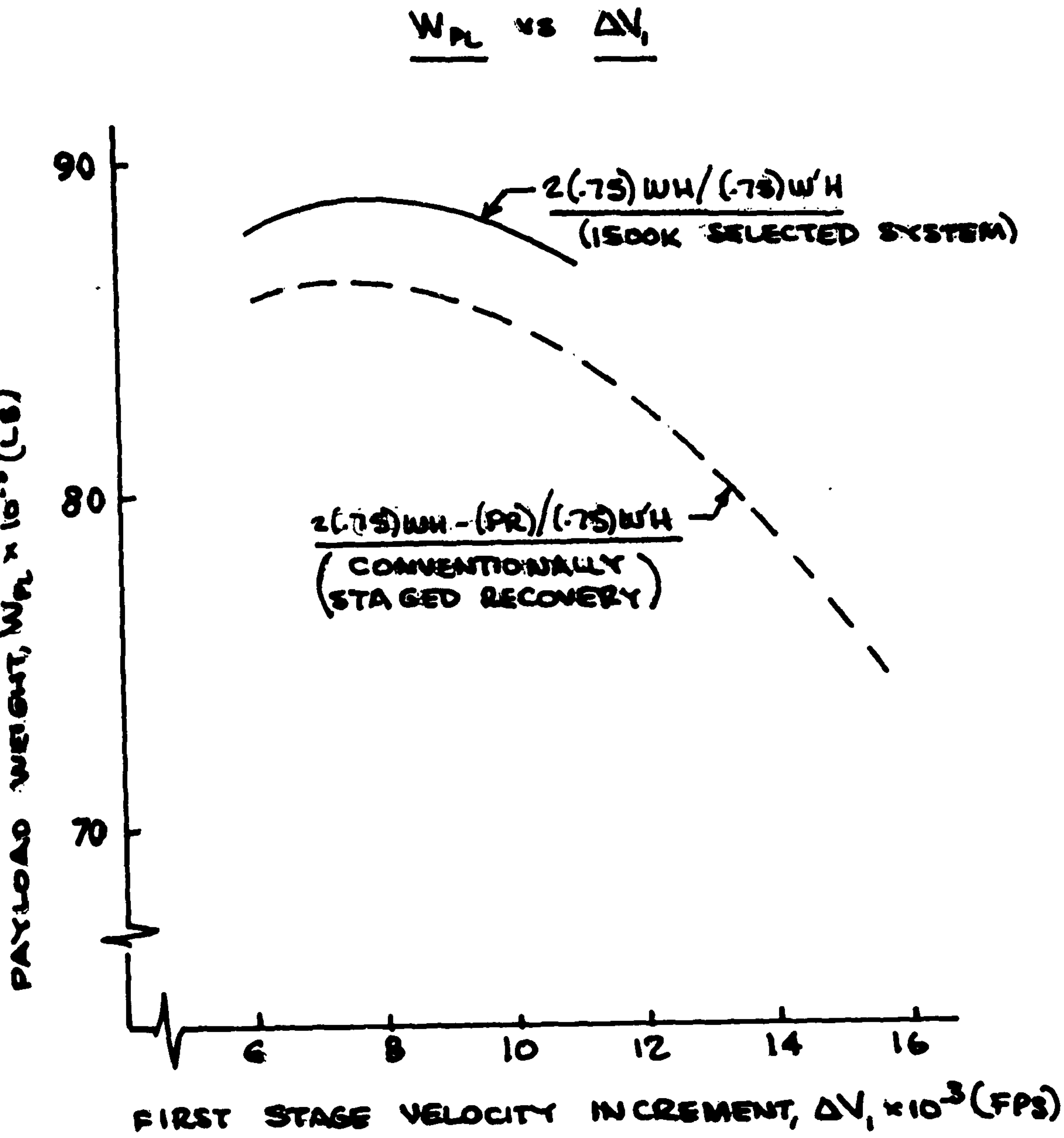


Figure 45. Payload Weight Versus First-Stage Velocity Increment for 2(0.75)WH/0.75 W'H - Effect of Paraglider Recovery

DYNAMIC CONSIDERATIONS

The subjects to be discussed in this section are separation, vehicle bending, propellant sloshing, and staging. The information presented in this section has been excerpted from recent S&ID studies on large booster systems in the thrust range of interest. This has been done because (1) detailed analyses of dynamic considerations for the myriad configurations discussed in this report was clearly impossible within the established funding, (2) the objective of the Cost Optimized Booster Systems Study was interpreted to be the investigation and discovery of new and promising large booster systems and most of the original work was expended in performance and costing evaluations, and (3) the work previously completed on dynamic considerations for large booster systems was representative in nature and was perfectly valid for the configurations investigated in this report.

SEPARATION

The separation of two booster stages is a complex process characterized by split-second action between moving surfaces whose clearances are measured in inches. Disturbing forces of considerable magnitude may be encountered and are derived from transients during the lower-stage engine shutdown, the upper-stage engine start-up, aerodynamic forces, initial rotations, and ullage and retro-rocket misfiring and misalignment. Positive acceleration must be maintained on the upper stage during light-off of the upper-stage engine in order to maintain the propellants in the correct location in the tanks. A separation sequence may be determined only after due consideration of all of these factors, plus consideration of the effect of various solutions on vehicle performance.

A detailed study has been made of the dynamics of the separation of a small upper stage (330,000 pounds of propellant) from a 3000K booster. A separation condition which resulted in a dynamic pressure of 287 psf was assumed in this study. The disturbances assumed were as listed in Table 10 below, and were combined in the most adverse manner.

Table 10. Stage Separation Disturbances

SOURCE	MAGNITUDE
First Engine Shutdown	±1/2 percent thrust variation per engine
Second Start-up	Normal or one engine out
Retro and Ullage Rockets	All misaligned 1/4 degree in same direction. One out of eight fails to ignite
Aerodynamic Forces and Initial Rotations	Obtained from trajectory with 3σ wind inputs

During the study, the following five modes of recovery were investigated:

1. Fire-in-the-hole - The separating force is supplied by the second-stage thrust and the reaction of the jet on the first stage. Ullage and retro rockets are not required. The J-2 mainstage signal is given just prior to separation.

2. Drag flap - Drag brakes formed by splitting the interstage structure are used to replace the retro rockets. Maximum interstage structure is jettisoned. Maximum structural clearance is provided for the second-stage engines. Ullage rockets on the second stage are avoided by sequencing main stage at separation.

3. Coast - The second-main-stage signal is not given until after the nozzle has cleared the first-stage structure. Ullage and retro rockets are used to provide the necessary accelerations.

4. Sliding interstage - The interstage slides back over the first stage immediately after separation. Main stage is then signaled. Retro rockets are used to slide the interstage back.

5. Interstage removal after separation - Interstage is jettisoned from stage two after separation. It slides along rails to prevent interference with second-stage engines. Retro rockets and ullage rockets are required.

A summary of the relative advantages and disadvantages of each are presented in the Table 11. A numerical rating system of 1 to 5 indicating a range from "best" to "worst" for each parameter is used.

Table 11. Separation Mode Comparison

	Coast	Fire-In-The-Hole	Drag Flap	Inter-stage Removal	Sliding Inter-stage
	1.6 per-cent	0.20 per-cent	0.5 per-cent	1.9 per-cent	0.8 per-cent
Payload Weight Penalty	4	1	2	5	3
Control simplicity	3	1	2	5	2
Sequencing sensitivity	1	5	1	1	1
First-stage recovery	1	4	5	1	3
Engine-out compatibility	1	5	1	1	1
Vehicle divergence	5	2	1	3	4
Explosion hazard	1	5	4	2	2
Aerodynamic sensitivity	5	1	4	2	2
Overall system simplicity	1	2	5	3	4

The coast mode is very good except in vehicle divergence due to the length of time without thrust vector control. The weight penalty is high because of the amount of interstage structure and the ullage rockets on the second stage. Fire-in-the-hole is generally poor operationally with extreme sensitivity to timing. The drag flap mode has merit but is complicated. Interstage removal has a high payload penalty and requires complex control; however, the explosion hazard is low as is the sensitivity to sequencing. The sliding interstage is complicated and has high vehicle divergence. Note that wherever the explosion hazard is high, recovery of the first stage is a poor probability.

An examination of separation conditions for the various vehicles indicates that the dynamic pressure will be between 30 and 100 psf. This reduction in air loads from the conditions investigated makes the coast mode of separation the most attractive because the divergence will be considerably reduced.

VEHICLE BENDING

Potential stability problems are always present in large booster systems. Due to the large weight penalties involved, the design cannot be based upon the stiffness requirements set by decoupling of the control system and flexible body frequencies. The large variations in vehicle weights, moments of inertia and center of gravity during the course of the trajectory due to the consumption of propellants creates a large variation in vehicle dynamic behavior. These variations impose certain limitations on the flight control system as it must provide closed-loop stabilization of the vehicle throughout its trajectory.

Preliminary analyses of the flexible body-control system coupling problem have been conducted recently by S&ID to determine the magnitude of complexity which would be required to stabilize the S-II stage of the Saturn C-3 escape configuration. This booster system is typical of the 600K to 3000K vehicles considered in the Cost Optimized Booster Systems Study. These Saturn studies demonstrated that with proper pitch-rate gyro location along the fuselage the destabilizing effect of body flexibility could be minimized (Reference 8).

The basic stabilization problem arises as control sensing elements, such as attitude position and rate gyros, pick up the local fuselage bending rotations and vehicle rotations. The placement of these sensors becomes critical due to the coupling of the control system with the flexibility modes. This is demonstrated by constructing a locus of the open-loop zeros of the longitudinal closed-loop control system, for example. The attitude position gyro is assumed to be placed in the payload and its position is moved along the body of the vehicle. The rate sensor is much more sensitive to the high frequency flexibility vibrations; therefore, its position is most critical. The bending frequencies are lowest at the start-burn condition and most likely to couple with the control system.

Analog studies indicate that a proportional control system with linear feed-back can be adequately stabilized provided the attitude-rate sensing device is carefully located. Further study will be necessary to define the tolerances allowable in the modal data for the particular rate-sensor location.

If a desirable rate-gyro location does not exist, or if it is impractical to place the rate gyro at such a station, the alternative is a phasor-cancellation control.

FUEL SLOSHING

The large amounts of fluid propellants aboard a large booster complex create yet another stability problem. Sloshing of these fluids produces lateral forces which are large enough to couple with the vehicle closed-loop system. Dynamic instability may result from this coupling, which must be eliminated through network compensation via the control system or by use of a baffled-tank construction.

Previous large booster studies conducted by S&ID have shown that fuselage flexibility and fuel-slosh dynamics do not intercouple provided that the lowest bending modal frequency is larger than the highest slosh-modal frequency by a factor of three or greater. The orthogonality of modes was affirmed in the present study. For this reason, the fuel slosh and fuselage flexibility sections of this report are presented separately for clarity to the reader.

The general methods presently in use at S&ID for propellant-slosh analyses are contained in References 1 and 2 and are verified experimentally in References 3, 4, and 5. Since it is realized that fluid-slosh dynamics is a relatively new field, effort is continually being made to refine these methods to provide a more accurate evaluation of the effects of propellant slosh on vehicle behavior. The analysis method used assumes that fluid-rotary motion is negligible. Rotary baffles should be placed in each tank to ensure that this motion is minimized.

The methods used for flexibility analysis are presented in Reference 7. Solutions to the partial differential equation for a non-uniform beam are obtained in the form of orthogonal modal functions of time and distance along the beam.

A general perturbation equation matrix combining fuel slosh and fuselage flexibility dynamics is presented in Figure 46. This matrix applies to any arbitrary number of slosh and flexibility modes.

A recent S&ID slosh analysis of a 3000K booster system is contained in the propulsion section of Reference 8.

PARALLEL STAGING VERSUS TANDEM STAGING

The staging mode considered for the large booster systems in this Cost Optimized Booster Systems Study with one exception has been from

PERTURBATION EQUATION	$\Delta\alpha_p$ (DEGREES)	$\Delta\theta_m$ (DEGREES)	Δq_{i_p} (FEET)	$\Delta(\Delta_{oj_p})$ (FEET)	$\Delta(\Delta_{Fk_p})$ (FEET)		$\Delta\beta_p$ (DEGREES)
NORMAL ACCELERATION	$-(S)-\frac{T}{mV}-\frac{57.3\bar{q}S}{mV}C_{N_{\alpha_p}} + \frac{g}{V}\sin(\theta_m-\alpha_p)$	$(S)-\frac{g}{V}\sin(\theta_m-\alpha_p)$	$-\frac{57.3\bar{q}S}{mV}\left[C_{F_{q_p}}-\frac{57.3}{V}C_{F_{\dot{q}_p}}(S)\right]-\frac{T}{mV}\lambda'_{q_p}$	$-\frac{57.3}{mV}K_{o_p}$	$-\frac{57.3}{mV}K_{F_p}$	=	$-\frac{T}{mV}$
PITCHING ACCELERATION	$-\frac{57.3\bar{q}S\bar{c}}{12 I_{yy}}\left(\frac{X_{cp}-X_{cg}}{\bar{c}}\right)C_{N_{\alpha_p}}$	$-(S)^2+\frac{57.3\bar{q}S\bar{c}}{12 I_{yy}}C_{m_q}\frac{\bar{c}}{24V}(S)$	$\frac{57.3\bar{q}S\bar{c}}{12 I_{yy}}\left\{C_{m_{q_p}}-\frac{57.3}{V}C_{m_{\dot{q}_p}}(S)\right\} - \frac{(57.3)T}{I_{yy}}\left[\phi_{q_p}+\frac{\ell_e}{12(57.3)}\right]\lambda_{q_p}$	$-\frac{57.3}{I_{yy}}\left(\frac{X_{cg}-X_{o_p}}{12}\right)K_{o_p}$	$-\frac{57.3}{I_{yy}}\left(\frac{X_{cg}-X_{F_p}}{12}\right)K_{F_p}$	=	$-\frac{T\ell_e}{12 I_{yy}} - \frac{4}{I_{yy}}\left\{\left[\frac{\partial M_R}{\partial\beta_p}\right]'-\frac{\ell_R}{12}\left[\frac{\partial F_R}{\partial\beta_p}\right]'\right\}$
GENERAL i^{th} BENDING MODE DEFLECTION FUNCTION	$\frac{\bar{q}S}{m}C_{Q_{i\alpha_p}}$	$\frac{\bar{q}S}{mV}C_{Q_{i\dot{\theta}_m}}(S)$	$-\left[(S)^2+2\zeta_{q_i}\omega_{q_i}(S)+\omega_{q_i}^2\right] + \frac{\bar{q}S}{m}\left[C_{Q_{i q_p}}-\frac{57.3}{V}C_{Q_{i\dot{q}_p}}(S)\right]+\frac{T}{57.3m}\phi_{q_i}\lambda_{q_p}$	$\frac{K_{o_p}\phi_{i(X_{o_p})}}{m}$	$\frac{K_{F_p}\phi_{i(X_{F_p})}}{m}$	=	$+\frac{1}{57.3}\frac{T\phi_{q_i}}{m} - \frac{4}{m}\left\{\left[\frac{\partial M_R}{\partial\beta_p}\right]'\frac{\lambda_{q_i}}{57.3}+\left[\frac{\partial F_R}{\partial\beta_p}\right]'\right\}$
GENERAL j^{th} MODE OXIDIZER SLOSHING MASS DEFLECTION	$-\frac{\bar{q}S}{m}C_{N_{\alpha_p}}$	$\left(\frac{X_{cg}-X_{o_j}}{12(57.3)}\right)(S)^2+\frac{V}{57.3} - \frac{g\sin\theta_m}{57.3}$	$-\phi_{q_{i(X_{o_j})}}(S)^2-\frac{\bar{q}S}{m}\left\{C_{N_{q_p}}-\frac{57.3}{V}C_{N_{\dot{q}_p}}(S)\right\} - \frac{T}{57.3m}\lambda_{q_p}$	$-\left[(S)^2+2\zeta_j\omega_j(S)+\omega_j^2\right] - \frac{K_{o_p}}{m}$	$-\frac{K_{F_p}}{m}$	=	$-\frac{T}{57.3m}-\frac{4}{m}\left[\frac{\partial F_R}{\partial\beta_p}\right]'$
GENERAL k^{th} MODE FUEL SLOSHING MASS DEFLECTION	$-\frac{\bar{q}S}{m}C_{N_{\alpha_p}}$	$\left(\frac{X_{cg}-X_{F_k}}{12(57.3)}\right)(S)^2+\frac{V}{57.3}(S) - \frac{g\sin\theta_m}{57.3}$	$-\phi_{q_{i(X_{F_k})}}(S)^2-\frac{\bar{q}S}{m}\left\{C_{N_{q_p}}-\frac{57.3}{V}C_{N_{\dot{q}_p}}(S)\right\} - \frac{T}{57.3m}\lambda_{q_p}$	$-\frac{K_{o_p}}{m}$	$-\left[(S)^2+2\zeta_k\omega_k(S)+\omega_k^2\right] - \frac{K_{F_p}}{m}$	=	$-\frac{T}{57.3m}-\frac{4}{m}\left[\frac{\partial F_R}{\partial\beta_p}\right]'$

LONGITUDINAL SHORT PERIOD PERTURBATION EQUATIONS

NOTE: SUBSCRIPTS p TAKE ON ALL VALUES OF i IN THE BENDING COLUMN, ALL VALUES OF j IN THE OXIDIZER COLUMN, AND ALL VALUES OF k IN THE FUEL COLUMN.

Figure 46. Longitudinal Short Period Perturbation Equations

conventional tandem configuration. The advantages of this proven method include low-aerodynamic drag losses, low-interstage connecting weight, no scars remaining on the advancing stage, lack of shockwave interaction with the advancing stage, and minimum danger of collision between advancing and retreating stages. The primary difficulty with the tandem-staging scheme is the long, whippy first-stage configuration which aggravates the already difficult problem of controlling a large booster system. The difficulty arises because the natural frequencies of the primary body-bending modes of large booster systems are low. Tandem staging increases the difficulty by providing a high first-stage fineness ratio, which further reduces the bending frequencies. The tandem configuration presents more problems than the parallel or laterally-staged configuration, particularly during the early flight period.

The advantage of the parallel staging technique was to dispose of dead weight more rapidly than would be possible with the tandem configuration. As the propellant was used from the first stage of a tandem two-stage booster, more and more of the first-stage tankage structure became superfluous. However, this now excess structure had to be accelerated throughout the first-stage boost phase until all of the structure could be jettisoned at first-to-second staging.

A suggested solution to the problem was to divide the equivalent of the first-stage propellant into an even number of peripheral tanks surrounding the second stage, to top the second-stage tank (which mounted all of the engines) from one pair of peripheral tanks at a time, and to jettison the empty pair as soon as the propellant had been transferred. Symmetry could be maintained by dropping diametrically opposed pairs of tanks. Practical considerations limited the maximum number of pairs to about four.

The laterally-staged configuration was the object of an extensive study by the Douglas Aircraft Company in an attempt to improve the effective propellant mass fraction of rocket-powered vehicles (Reference Douglas Report JM-38844). The study results were that a nine-tank, four-drop vehicle was best based on the computation of straight impulsive velocity. When aerodynamic drag and gravity losses were considered, the best configuration used three drops. When clustering penalties were taken into account, the two-drop vehicle was optimum and, finally, when reliability was introduced the one-drop, three-tank vehicle was the best. Douglas concluded that the one-drop configuration was competitive with tandem two-stage vehicles when considering the combination of performance, reliability and cost. By dropping 50 percent of the engine weight it was shown that a payload increase in the order of 7 percent could be achieved.

The results of the Douglas study seem to indicate that further investigation of parallel-staging concepts is desirable. The study would be limited

to one-drop, three-tank configurations. Costing, performance, systems and reliability factors should be determined for the one-drop vehicle which disposes of 50 percent of the start-burn engine weight because an increase of 7 percent in payload is appreciable. Studies of permutations of this staging scheme should prove equally worthwhile. Examples of permutations are to mount the droppable engines on the peripheral tanks. The peripheral tanks would feed the droppable engines and simultaneously top the second-stage tank; or the peripheral engines could be short-burning solid propellants to provide incremental thrust until the second-stage T/W reaches a desired level.

It should be noted that the discussion presented here is concerned with increasing the performance of a booster stage by supplementary peripheral tanks and engines. The technique is to get the stage, whose engines have been burning from launch, to high altitude and velocity with a full or nearly full fuel tank. The derivation of an optimum configuration has not been obtained either in the Douglas report or in this report.

FLIGHT CONTROL

The flight control system of a large booster missile will provide the forces and moments necessary to stabilize and control the missile along its predetermined trajectory in response to command signals usually originating in the payload. This Cost Optimized Booster Systems Study is too broad to present detailed considerations of all of the control systems which would be required. However, S&ID has completed extensive studies of large booster control systems and many specific recommendations can be made.

Potential stability problems are always present in large booster systems. Due to the weight penalties which would be involved, the structural design cannot provide the stiffness required to decouple the control system and flexible body frequencies. The large variations in vehicle weight, moments of inertia, and center of gravity during boost due to the consumption of propellants create correspondingly large variations in vehicle dynamic behavior. These variations impose limitations on the flight control system since it must provide closed-loop stabilization of the vehicle throughout the trajectory. The basic stabilization problem arises because the control sensing elements, such as attitude-position gyros and rate gyros, are sensitive to the local body bending rotations as well as vehicle rotations. The placement of these elements is extremely critical and must be determined with care for each configuration.

Control nozzle deflections and nozzle rates required will be determined by separation conditions. S&ID studies have indicated that relatively long time periods (in the order of 5 seconds) will be required between separation and the attainment of full thrust on the second-stage vehicle. The long coast period is required to provide adequate separation between stages and to allow sufficient time for thrust build-up on the second stage. The alternative to a long coast period is added mechanical and system complexity in the form of blast doors in the interstage structure. The maximum deflections and deflection rates will be required to control the separation transients and the divergence build-up during the uncontrolled period of separation. Preliminary separation studies indicate that deflection rates of less than 10 degrees per second will be required. This is well within the present state-of-the-art; possible adaptation of current developmental hardware for the Rocketdyne H-1 engines would allow a rate of 20 degrees per second for the J-2 engines, for example.

The positioning accuracy required for the movable nozzles will be about 0.25 degrees. The accuracy requirement is dictated by orbital trajectory considerations.

ATTITUDE CONTROL REQUIREMENTS

For conventional bell-nozzle engines, attitude control forces are best provided by gimbaling the engines. Each engine can be provided with two actuators, one mounted in the vehicle's pitch plane and one in the yaw plane. For engines the size of the Rocketdyne F-1, four degrees of pitch and yaw deflection is a reasonable state-of-the-art number. For J-2 engines the pitch and yaw deflections can be increased to 7 degrees. In the latter case, sufficient control is available for configurations with four or more engines to handle either an engine-out condition or a failure of the actuators which drives one nozzle hard over. The lower deflection limits associated with F-1 size nozzles allow for control of engine failure or actuator failure in one of eight or more engines. The use of pitch and yaw actuators on each nozzle in groups of two or more nozzles permits three-axis stabilization and control. Roll control for single-engined configurations requires an auxiliary system and is discussed in a subsequent paragraph.

The development of a satisfactory control system for plug-nozzle engines will probably evolve with the engine designs. For the present, pitch and yaw attitude control can be assumed to be obtained by either secondary injection methods or by throttling portions of the nozzles. Either method produces an asymmetric exhaust pattern and consequent bias of the thrust vector. The only plug-nozzle configuration considered in this study has a single-thrust chamber and, therefore, a secondary control system will have to be employed to provide roll control.

The problem of a satisfactory flight control system for the Pratt & Whitney advanced high-pressure engine powered single-stage-to-orbit configurations is complicated by the fact that the nozzles cannot be gimbaled due to mechanical considerations. The current suggestion is to provide secondary-reaction jets in a cross section plane near the high-pressure combustion chamber stations and to power the reaction jets with gases tapped from these chambers. Aerodynamic surfaces may be employed in connection with the reaction jets to provide most of the control forces required while the booster is in the sensible atmosphere. Canard surfaces in connection with aft fins to provide aerodynamic stability are particularly effective. Weight trade-offs must be conducted to optimize the control system when multicontrol systems are contemplated. An auxiliary aerodynamic control system has been sized for the single-stage-to-orbit configuration and is presented in Figure A-3.

ROLL CONTROL REQUIREMENTS

In the case of large booster missiles powered by multiengined conventional bell-nozzle systems, roll control can be obtained differentially deflecting the nozzles. For single-nozzle, plug-nozzle, and other fixed

nozzle configurations, roll control must be obtained by auxiliary systems. One method is to use auxiliary reaction jets fueled by either a separate propellant system, by the primary propellant system, or by gases tapped from the combustion chambers of the primary engines. Aerodynamic surfaces may be used to supplement or to complement the auxiliary jet system while the missile is in the atmosphere. Final selection of the type of the roll control system will depend on sizing and weight trade-off considerations.

RECOVERY ANALYSIS

This section presents a generalized discussion of recovery techniques as applied to cost-optimized booster systems. Substantiation of the economic advantage of recoverability will be given in a subsequent section. The advantages and disadvantages of three-unique recovery systems are presented in addition to suggested configurations based on previous studies.

For the purpose of this study, only land recovery was considered. Sea recovery by modified aircraft carriers or similar seagoing vehicles appears to have too many problems in the fields of guidance, cost, landing control, and general operational feasibility to merit consideration.

RECOVERY CONCEPTS

Parachute

Parachute recovery offers the simplest and most proven means of recovering payloads reentering the atmosphere at hypersonic speeds. When extended to booster recovery, however, several problems become evident. One would be the limited range control because the number and location of downrange recovery sites, specifically from Cape Canaveral, would severaly limit azimuth and end-boost range values. Another would be that extensive shock attenuation devices would be required to minimize the probability of damage at touchdown. Lastly, the state-of-the-art advancement of parachute design as applied to such heavy loads would be expected to seriously lag booster design advancements; that is, such a large number of parachutes would be required that the system would be impractical and unreliable. To alleviate these shortcomings, attention was focused on flyable boosters.

Fixed Wing Recovery

From the standpoint of high reliability, fixed wing recovery offers a very promising means of booster recovery. A fixed wing recovery system was investigated and the results of the study were used to evaluate the concept as applied to cost optimized boosters.

The wing is a 60-degree leading edge sweep, symmetrical cross-section airfoil with a design wing loading of 35 pound per square foot. Pitch control is obtained by means of split-flap control surfaces mounted on the

wing trailing edges and extending outboard 70 to 80 percent of the wing span. Elevons are located between the flaps and wing tips for roll control. Stationary vertical surfaces are mounted at the wing tips to provide additional lateral stability and control surface effectiveness. The control surfaces are also used as drag flaps during reentry primarily for the purpose of decreasing wing root-chord shear forces during deceleration.

The primary disadvantage of the fixed-wing concept lies with the almost intolerable weight penalty and the inherent structure problems. It is estimated that the booster dry weight would be increased 30 to 40 percent by the addition of wings. Since the only advantage of fixed wing recovery over paraglider recovery appeared to be the slightly higher reliability, primary attention was focused on the lighter paraglider recovery system.

Paraglider Recovery

Extensive investigations have been made into paraglider recovery of large boosters as the Saturn S-1, C-2 and C-3 configurations shown in Ref. 9 and 10. The feasibility of such a recovery concept has been demonstrated not only by theoretical means but by considerable wind tunnel testing and flight tests. The necessity for a light-weight, reliable, and practical paraglider recovery system has encouraged considerable activity in the industry resulting in several unique configurations. This section will be devoted to outlining a generalized paraglider system and method of recovery.

PARAGLIDER RECOVERY OF CONVENTIONAL 600K BOOSTERS

Configuration

The major parameters affecting the paraglider configuration are wing size, type of structure, wing loading, and allowable keel length. The optimum wing shape is generally agreed to have the leading edge booms equal in length to the keel and oriented in a flat planform sweep angle of 45 degrees (55 degrees in the deployed condition). Folding spreader bars are employed to maintain the optimum aerodynamic wing geometry and reduce boom bending loads.

A design wing loading of 15 pound per square foot was recommended by both of the aforementioned studies and was tentatively selected for this configuration because of the amount of data available. This wing loading value also appears to be the minimum value for adequate aircraft tow capabilities (to be discussed in a subsequent section).

Knowledge of representative booster dry weights and the design wing loading allows the wing area to be sized and, hence, the keel length. The

tank length of the representative boosters was such that the required keel length did not exceed the allowable keel length from rocket nozzle exit to booster nose shoulder, thus eliminating the need for inflatable booms similar to Saturn Paraglider or folding rigid booms. The selection of a rigid nonfolding paraglider similar to the C-3 booster Paraglider was then made primarily on the basis of the simplicity and higher performance of the rigid-boom structure as compared to the inflatable-boom structure. A reliability and feasibility comparison should be made only after more hardware testing has been accomplished. Obviously, development costs of each must be taken into account prior to a final selection.

The control system of the C-3 booster-paraglider system may be described as a booster attitude control system with a fixed geometry between booster and paraglider. Booster pitch, roll, and yaw control is maintained by conventional split-flap control surfaces on four booster stabilizing fins. Due to weight considerations, this system was discarded in favor of the lighter control system of the Saturn Paraglider. In this system, control is obtained by varying the relative position of the paraglider to the overall center of gravity. The forward and aft keel shroud line lengths are varied to provide pitch control, and the leading edge boom shroud lines are proportionally altered to provide roll control. Once again, feasibility, development costs, and reliability will enter into the final control system selection.

Further design details, ground systems requirements, and substantiating data may be found by referring to Ref 9 and 10.

Method of Recovery

Subsequent to first-stage burnout and separation, an auxiliary drag device such as a balloon or drag chute is deployed. This serves the dual purpose of decelerating the vehicle to subsonic speeds at high altitudes and reducing the range from the launch point.

The paraglider is deployed in the subsonic Mach number region due primarily to the marginal stability of the booster-paraglider combination in the transonic-supersonic region. The concepts of deployment at either apogee or minimum dynamic pressure were discarded not only because of the stability considerations but because of the high-heating rates incident on the paraglider fabric and the steel cable shroud lines. The deployment sequence is similar to the method shown in Figure A-6.

After deployment, the paraglider pulls the booster out of the ballistic dive and the configuration seeks an equilibrium glide path angle. The range capability of the paraglider is then dependent on the lift-to-drag ratio of the paraglider and the altitude at which the glide path is attained. Due to

the rather close azimuth tolerance required to glide to one of the stations downrange from Cape Canaveral, the provision for tow capability of the booster was investigated. This capability also generalizes the recovery concept to any launch site.

Figure 47 presents the paraglider flight parameters pertinent to air rendezvous and tow. The overall lift-to-drag ratio is presented as a function of the paraglider keel angle of attack for various booster angles of attack. Equilibrium glide velocity is also presented for the nominal booster angle of attack ($\alpha = 0$ degrees). Since the rendezvous maneuver has not yet been completely defined, no attempt has been made to correlate booster attitude and velocity with the minimum cruise velocity and associated flight path angles of the designated tow aircraft. However, the data presented are considered adequate for defining the booster parameters during the rendezvous.

The equilibrium flight towing capabilities of the C-133A tow aircraft are demonstrated in Figure 48. The C-133A was selected on the basis of its low stalling speed and maximum low-speed performance margin. Level flight at an altitude of 10,000 feet is assumed. Also presented are the tow requirements of three boosters whose weights bracket those encountered in the 600K first-stage thrust study. It can be seen that the performance margin of the system allows the paraglider to be flown at off-optimum lift-to-drag ratios and resulting higher velocities for a more comfortable flight regime for the C-133A.

The major control system requirements are encountered during the flare maneuver between tow release and touchdown. Extensive analog investigations (See Ref. 9) have resulted in a control system concept which senses altitude and altitude rate ($\dot{h}$) for flare trajectory control. These parameters are compared with reference values for the generation of the feedback error signal. This error signal is then used as a vernier correction to an open-loop program of cable length control for the flare maneuver. Further system description and the effect of parametric variations in the control system and the flight regime may be found in Ref. 9.

Recovery of larger thrust-level vehicles is very nearly identical in method and configuration with one exception: the thrust requirements of the tow aircraft increase almost proportionally with the first-stage thrust level. Figure 49 presents a preliminary estimate of the thrust required to tow the first-stage booster of an optimum 1500K configuration. Also presented is an estimate of excess thrust available with a C-133A modified with two J-79 turbofan engines. Justification of J-79 engine addition to the standard C-133A is given in Ref. 9.

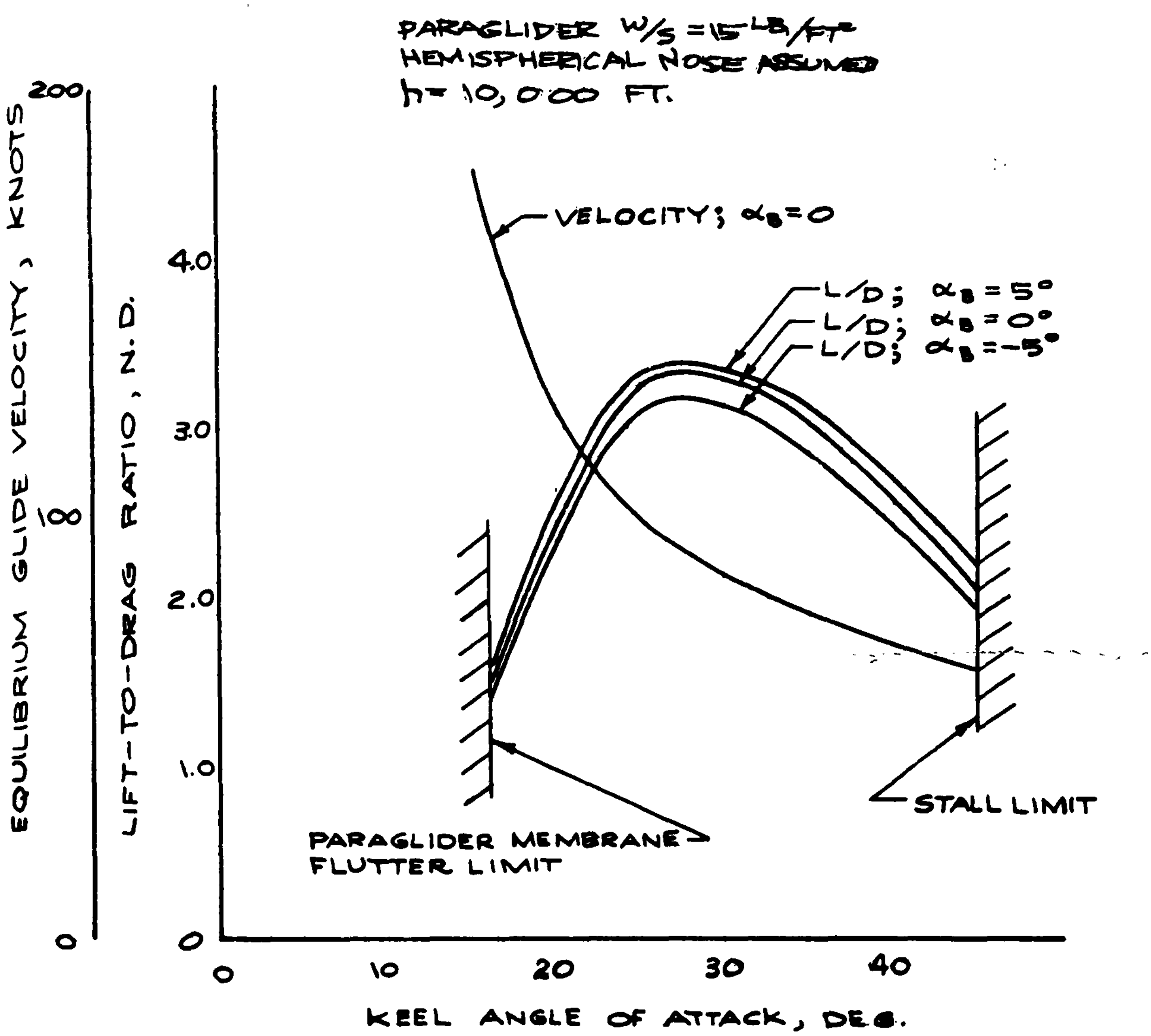


Figure 47. Paraglider Performance During Air Rendezvous

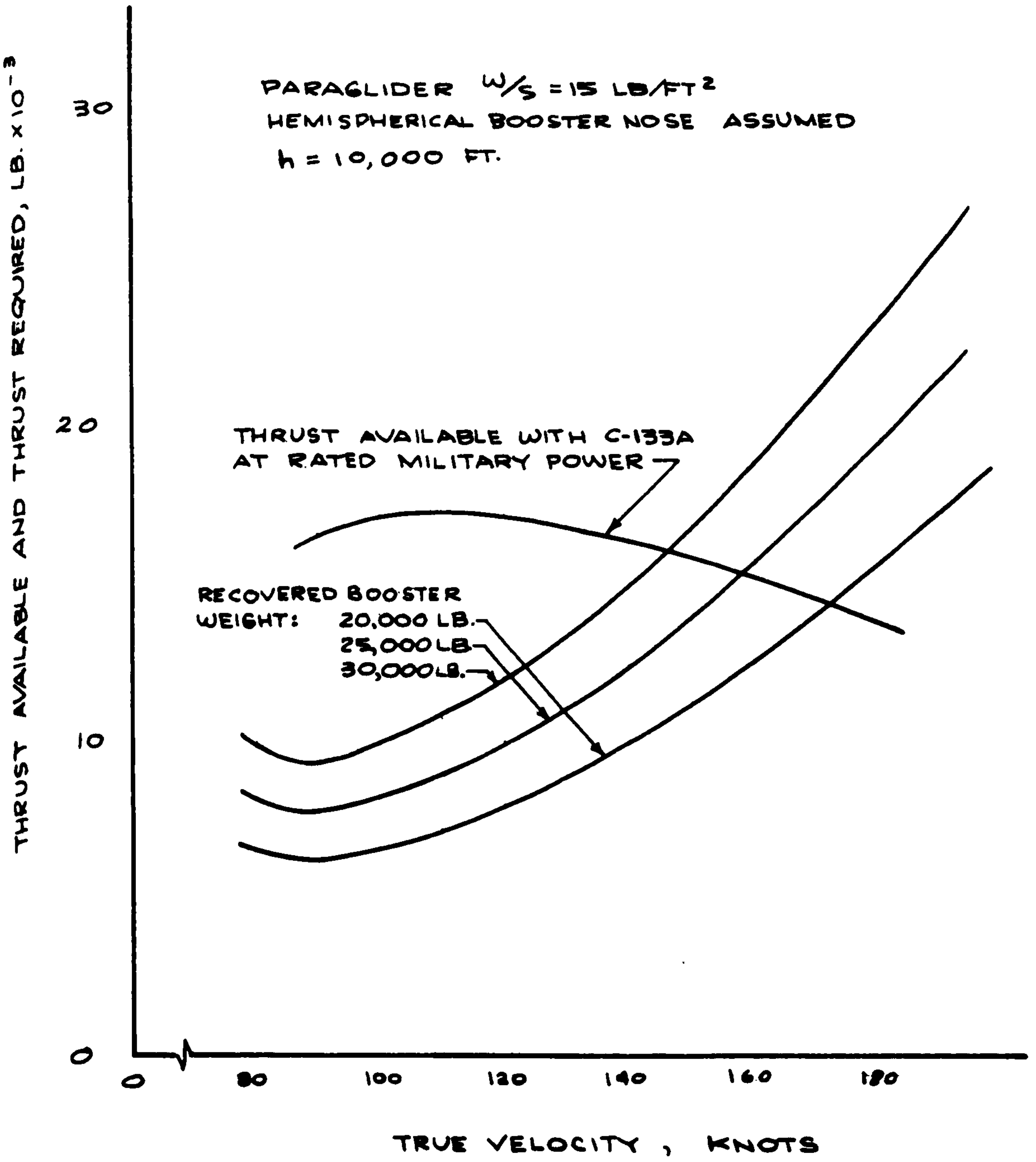


Figure 48. Towing Thrust Requirements for 600K Booster Recovery

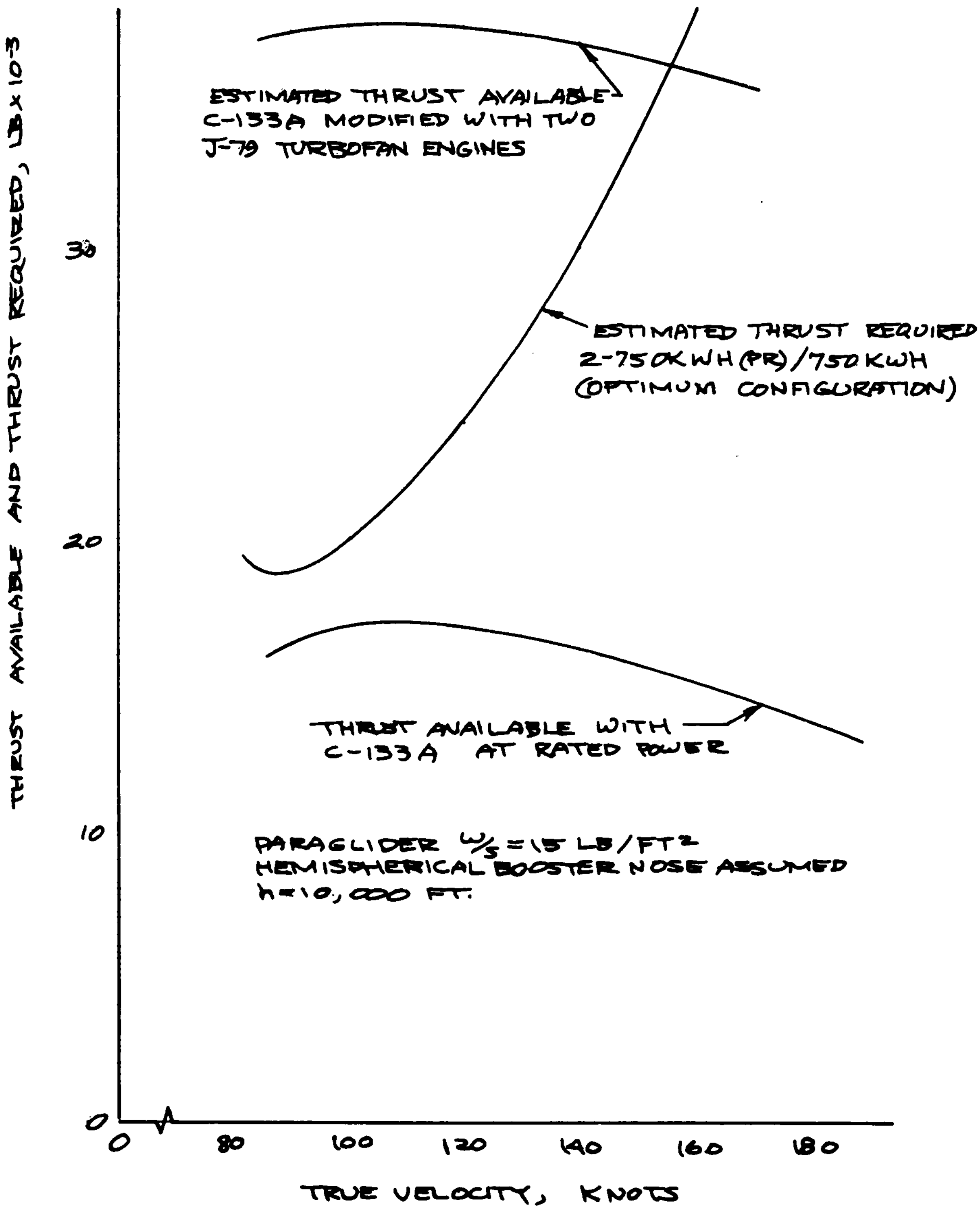


Figure 49. Towing Thrust Requirements for 1500K Booster Recovery

PARAGLIDER RECOVERY OF EXTENDED FIRST-STAGE BOOSTERS

The advantages of recovering off-optimum extended first-stage boosters can be found in an economic evaluation of the launch system. Extended 1500K boosters were not investigated, but since the weights of extended 600K first-stage boosters fall within the aforementioned brackets (Figure 48), no problems are evident in the actual mechanics of recovery.

Paraglider recovery of large boosters appears to be promising not only from a feasibility standpoint but also from a cost standpoint. Future analysis in the general field and application to current design boosters will necessarily decrease paraglider development costs and increase reliability and performance of the system concept. Extension of the concept to recovery of boosters larger than the 1500K example is limited by the capabilities of the tow aircraft.

CONCLUSIONS

The most promising design philosophies and vehicle configurations for the Cost Optimized Booster System Study have been selected.

The initial flight mechanics analyses selected stage optimized 600K, 800K, 1500K, and 3000K systems. In those vehicles where RP-fuel was used in the first stage and LH_2-fuel was used in the second stage, the staging optimization favored the upper-stage propellant mass fractions. In general, the first-stage velocity increment for the LO_2-RP/LO_2-LH_2 system ranged from 4000 to 8000 fps whereas for an all liquid hydrogen system, LO_2-LH_2/LO_2-LH_2, the first-stage velocity increments ranged from 8000 to 14,000 fps. Predicated upon the type of second-stage engine (WH, PH, or JH), the optimized second-stage initial thrust to weight ratio was always between 1.5 and 2.0. For each of the four 600K, 800K, 1500K, and 3000K systems, the maximum payload capability was demonstrated by the use of the Pratt & Whitney advanced high-pressure engine cycle in both stages. Cost savings were foreseen by adoption of a lower-stage dual engine installation wherein each individual engine was identical with the upper-stage engine. This feature was incorporated into the evaluation of the various selected systems.

Based upon the selected systems of the above studies, the final portion of the Cost Optimized Booster Systems flight mechanics investigations considered various modular arrangements: the use of solid-propellant and liquid-fluorine systems, identical nozzle expansion ratios in both upper and lower stages, a variation in engine chamber pressure, a single-stage-to-orbit concept using secondary nozzle expansion, and recoverable first stages.

In the modular studies, two-stage boosters utilizing three-, four-, and seven-clustered modules in the first stage and a common module in the second stage were considered first. The seven-module arrangement was omitted from further study when it was shown that the additional first-stage thrust increment did not orbit a proportional increase in payload weight. The three- and four-common-module systems orbited payloads of 105,000 pounds and 140,000 pounds respectively. For the remainder of the modular studies, a staging optimized second stage was substituted for the former common module. By this modification, payload weights increased from the above values to 120,000 pounds and 160,000 pounds for the optimized three- and four-modular systems, respectively.

In order to match the 40,500 pound payload of the 600K selected system for the segmented solid first stage, a take-off thrust of 1.4×10^6 pounds was required. The second-stage propellant mass fraction, υ_2, optimized at 0.80, much higher than the 0.66 recorded for the selected 600K system.

By use of identical nozzle expansion ratios ($\epsilon_1 = 45 = \epsilon_2$), the payload capability of the selected 600K system was matched. This feature would seem to indicate a cost reduction if adopted, particularly in the common-module concept.

The effect of an engine chamber pressure variation was studied next. All prior analyses had used a chamber pressure value of 3000 psia. By going to 5000 psia, a payload weight decrease of some 1000 pounds was incurred. This reduction justified our original 3000 psia selection.

In the study of the liquid fluorine system, a 2(0.3) WH/ (0.2) YF configuration put into orbit a payload of 40,000 pounds. This is only 500 pounds less than that of the selected 600K system, 2(0.3) WH/ (0.3) W'H. The $(T/W_o)_2$ for the fluorine version optimized at a lower value than that for the 600K selected system, 1.41 compared to 1.90.

Through the use of secondary expansion, reasonable payloads were realized for the single-stage-to-orbit vehicles. Two concepts were actually investigated. One concept used a separate solid kickstage whereas the other employed a relightable first stage. Because of pressurization, chill-down, and evaporation losses amounting to 5000 pounds in the relightable version only, this version lost much of its attractiveness. Depending upon the first-stage structure mass fraction, υ_{b_1}, the separate solid kick-stage concept orbited from 20,000 to 30,000 pounds in payload. This was at least 5000 pounds more than the relightable version as discussed above and ranged up to 5700 pounds more at the lower υ_{b_1} values.

Finally, from the first-stage recovery investigations it was seen that a 600K conventionally staged paraglider recovery system reduced the payload weight of the selected 600K system by 1000 pounds down to approximately 39,500 pounds. A similar 1500K paraglider version reduced the payload weight of the 1500K selected system from 90,000 pounds to 86,400 pounds, a decrease of 2600 pounds. The effect of extending the first-stage staging point and then recovering the first stage was seen to degradate the orbited payload capability still further. In spite of these performance penalties, the concept of paraglider recovery appears to be a solution to the ever-rising booster costs contingent upon the launch rate desired.

VEHICLES

SUMMARY

The various vehicle configurations have been generated to support the cost optimization program as previously outlined, and to delineate problem areas which are associated with specific types of engines, propellants or systems. All of the configurations shown in the appendix are based on a 300-nautical mile circular orbit requirement. The primary systems configured were two-stage, single-tank booster systems. In some cases, the second stage has been optimized for maximum performance. In others, an attempt was made to standardize the vehicle from a component standpoint; i.e., a method of trade-off was sought between maximum efficiency and lowest cost of the unit. In the alternate booster concepts, the machine optimization program was used where applicable only to establish the vehicle weights because of their peculiar design variations. These designs also were based on requirements to achieve a 300-nautical mile orbit. In the case of the lateral stage designs, the North American Aviation nine-engine concept has been compared to the single-plug engine Douglas design from the standpoint of cost per pound of payload in orbit. All of the machine-compared two-stage booster systems have been based upon LO_2/RP or LO_2/LH_2 for first-stage propellants and LO_2/LH_2 for second-stage. Alternate booster systems also included liquid fluorine oxidizer in a second-stage design and segmented solids for first stage.

In support of the cost optimization program, the geometry of the structure is of utmost importance in determining vehicle weight; therefore, simple outline drawings have been presented for this purpose. Consistent methods of material selection, fabrication methods, and assembly techniques have been adhered to in the designs. The type of structure was changed from a semimonocoque aluminum alloy only when it was found that the construction materials were incompatible with the propellants. In cases where a thermal expansion could cause difficulty in the connectors between tanks, the problem has been pointed out but no attempt was made to find the solution in the study. The designs presented for the machine program are the result of a single iteration in most cases; that is, after the second stage thrust level was established, and configuration was developed
important ground rule adhered to in supporting the program was consistency in the details of engine mount structure as well as propulsion systems. In all configurations, the fuel is located in the upper tank and the oxidizer is placed at the low level. The engine mounts have been standardized by using

a single cone or double cone, as the situation required. In cases where more than three engines were used, the standard engine mount ring was selected as the design. To support the vehicle while it is on the launch pad, it was assumed that actuated arms on the ground equipment could attach to the bottom of the tank and thus impose no scar weights for this purpose on the booster itself.

As the program continued, it became apparent that with different types of engines it was no longer possible to adhere to the previously used types of construction and methods of pressurization. In the high-pressure engine, if the gimbal point was located at the forward end of the engine, the overhanging moment would seriously penalize the engine as well as the overall configuration due to its high-fineness ratio. Therefore, as part of the problem areas in the study, methods were devised whereby a long engine, such as the high-chamber pressure design, could be mounted and gimbaled without requiring excessively heavy turbine case structure. The solution to this problem was integrated with considerations for propellant feed lines as well as flame shielding in the base area.

TWO-STAGE BOOSTER SYSTEMS

Major design emphasis was placed on configurations with thrust levels of 600K and 1500K pounds. The 800K and 3000K configurations were predicated upon the optimized 600K and 1500K systems, respectively, in order to keep the effort within the scope of the study. The configurations with advanced engines for various thrust levels are shown in Figure 1. The configurations with state-of-the-art engines are shown in Figure 2.

One of the basic inputs required before the weights mechanization program can proceed is the definition of the geometry of the vehicle to be analyzed. The drawings presented in the appendix provide the necessary dimensional definition of the two-stage tandem vehicles which were compared and used to determine maximum payload efficiency. Various types of engines and propellants were used in the comparison. Emphasis was placed upon consistency of criteria used when each different successive design was started. Any design improvements which naturally occur in the evolution process were kept out purposely so that the last configuration would reflect no advantage due to this learning.

In an effort to adhere to this design consistency, the entire mounting problem has essentially been segregated as a separate problem. As part of the engine mounting study, the problems and advantages of gimbaling were included and are discussed in the following pages. Thus it was possible, for example, to use a single type of engine mount for all designs which used two Pratt & Whitney engines.

All vehicles considered in the machine comparison program used gimbaled engines for thrust-vector control. In cases where only a single engine was required, the roll-control system was included in the weights analysis.

For support on the launch pad, all vehicles are assumed to react the forces uniformly. Separation is accomplished by primacord which encircles the skirt near the launch pad footings.

ALTERNATE BOOSTER SYSTEMS

Alternate concepts studied, as shown in Figure 3, were:

1. 600K first-stage clustered modules with common second-stage module and cost optimized second stage.
2. Paraglider recovery for 600K and 1500K conventional first stage and 600K extended first stage.
3. 600K segmented solid first stage with selected cost optimized second stage.
4. 200K liquid fluorine/liquid hydrogen cost optimized second stage on selected 600K liquid oxygen/liquid hydrogen first stage.
5. 1500K lateral stage system, including Douglas Aircraft common tankage concept and NAA common engine concept.
6. 600K single stage to orbit employing advanced high-pressure engine with secondary expansion nozzle.

Most of the alternate concepts were not applicable to the performance-weight-cost mechanization study; hence, the program was used only to establish weight of their peculiar design variations.

600K FIRST-STAGE CLUSTERED MODULES

The module employed for this study was based upon the 600K high-pressure engine first-stage booster system. The common module concept, which employed a single identical module for the second stage, is illustrated in Figure A-7. The structure criteria for each module is set by the second stage in these designs, and the second stage is mated with three, four, and seven identical units in the first stage. Figure A-8 illustrates similar arrangements of a three- and four- module first stage, except that the module thrust level is 1500K.

The optimized second stage on a modular first stage does not impose second-stage structural penalties on the first stage design. Coupled with the latitude of different second-stage thrust levels and tank size, this concept will orbit higher payloads due to higher performance mass ratio for a given first-stage thrust as compared to common module concept.

Clustering is accomplished by keeping the structure loads as symmetrical as possible and leaving minimum scar weights in the tanks. The basic engine shown in Figure A-7 is the 300K thrust version of which there are two on each module. A single 600K thrust engine could replace this propulsion system and would involve changing from a double conical thrust mount structure to a single cone and adding auxiliary rockets for roll control.

The skirt at the top of each module is designed to provide for the adaptor when the tanks are clustered, as well as support the payload when this module is used in the second stage. The connection between the cluster of tanks is undefined in detail, but is presumed to be of a continuous nature since the unit loading would be very low and thus impose minimum scar weights in the cylindrical structure of the tank. This is possible because, due to the more severe loading conditions imposed during second-stage operation, there will generally be a greater margin of safety when these tanks are used as first-stage components. Provisions for differential thermal expansion between tanks also must be provided.

FIRST-STAGE PARAGLIDER RECOVERY

To support the cost optimization studies, which were made to determine the effect of recovering the first-stage booster, three configurations were evaluated, all of which have provisions for a paraglider type of recovery on the first stage. Since it is necessary with this system to include a drag chute, the provisions for this container also have been shown. In addition, the performance of a paraglider on an extended first-stage booster has been evaluated and a comparison included in the cost analysis section. The extended first stage imposes the additional problem of recovery when higher boost velocities are used.

Minimum cost trends are established by trading size of the recoverable first-stage booster against size of the second-stage booster.

The three configurations provide the basis for analyzing the structure and consequently the performance and cost of each design. From this, the shape of the cost curve can be determined and the minimum booster cost plotted against launch rate. The total weight in all cases is to remain constant. However, since paraglider weight has been set as a fraction of the first-stage booster weight, the recovery system will increase as the size of the first-stage booster increases. To simplify the analysis, all paraglider systems were assumed to be 16 percent of the first-stage booster empty weight. The method of supporting the deployed paraglider inflatable structure is shown in Figure A-6. The pitch and yaw controls are handled by reeling in and letting out various support cables.

To determine the point at which a recovery system yields the greatest benefits, the size of the booster has been varied from that required for maximum payload to a size larger than that required. As the size of the first stage increases, it is necessary to decrease the size of the second stage to hold identical gross weights in all cases. Configuration A of Figure A-5, which is the conventional two-stage system, shows the highest payload can be carried with a second stage which is larger than configuration B or C. As the size of the first stage increases (and the size of the second stage decreases), the vehicle performance is degraded and the payload is reduced. However, this trend does not necessarily tell which configuration will yield the lowest cost per pound of orbital payload. To establish the point at which minimum payload cost will occur, the various size recoverable boosters were evaluated, and the results are shown in the Economics and Operations section. Essentially, an effort has been made to trade-off maximum staging efficiency against the cost of the first-stage booster. Since the cost of the system may be reduced by booster recovery, the three vehicles were designed to provide the geometry upon which the analysis was based.

A single recovery skid has been shown at the front of the booster and two skids are provided at the rear, since dry land recovery is considered to be most desirable from a minimum refurbishment standpoint. The skids are mechanically extended at the time of drag chute deployment and must be manually retracted before re-use.

In addition to the stage separation plane, which is located near the upper-stage tanks, a second point of separation has been provided near the first-stage booster. This method enables the drag of the vehicle to be reduced during the glide portion of the recovery procedure. To attain the lowest drag configuration, after the second stage has accelerated away from the booster, the interstage structure will be separated just ahead of the forward skid mounting frame. Separation of the interstage will occur after the drag chute has been deployed from the aft portion of the booster to prevent any physical contact between the two objects.

The design and analysis effort shown revolves about the rigid-boom paraglider concept shown in Figure A-4. The recovered booster illustrated in Figure A-6 is representative of a high density-type propellant (LO_2/RP) booster and is presented to illustrate deployment. It will be noted that the tank length is less than the length of the leading edge boom which is used on the rigid boom wing. In such a case, in order to obtain the required wing loading, (15-20 PSF), it is necessary to use a boom length greater than the length of the body.

Since the primary propellant considered in the study was the low bulk density LO_2/LH_2 it is unnecessary to use the inflatable boom to achieve the required length of the leading edge boom. Since the total length of the booster is greater in such designs, it is possible to store a rigid boom in a long container on the booster without provision for joints in the boom.

SEGMENTED SOLID FIRST STAGE

On the solid propellant first-stage booster shown in Figure A-9, it is necessary to hold a smaller diameter on the first stage than on the LO_2/LH_2 second stage due to the high-propellant density of the solid boosters and the very low density of the LO_2/LH_2 propellants. While the maximum diameter of the first-stage (solid) booster is limited, due to considerations which must be made to provide the proper grain casting shape and handling, this factor does not penalize the interstage structure. Essentially, the arrangement results in an interstage structure which is reversed from the versions where the diameter of the second stage is smaller than that of the first stage.

While this arrangement of masses poses no structure problems in the interstage, it does increase the control requirements of the first-stage booster due to the de-stabilizing effect of the comparatively large upper stage.

Roll control is accomplished by using dipper vanes in the exhaust stream of the first stage while attitude control is provided by a liquid injection system. This type of control system has been selected because of its adaptability to short burning time boosters such as the solid-propellant type. The system as shown has been based on a burning time of 72 seconds and uses nitrogen tetroxide stored in spherical bottles around the throat of the nozzle. To arrive at an initial estimate of the size of the control system, it was estimated that a normal force of 98,000 pounds would be required for a time period of 20 seconds to stabilize the vehicle at lift-off and to correct for wind shears during maximum dynamic pressure conditions.

LIQUID FLUORINE-LIQUID HYDROGEN SECOND STAGE

An LF_2/LH_2 system was studied for second-stage application on a 600K LO_2/LH_2 first-stage system. By employing the identical first stage used in a LO_2/LH_2 second-stage system, a direct comparison was possible. The configuration illustrated in Figure A-10 is based upon a 200K LF_2/LH_2 system which staging optimization studies showed to be optimum. By comparing this with the 600K all LO_2/LH_2 illustrated in Figure A-11, it can be

seen that the LF_2/LH_2 system is smaller, due to the higher propellant density of LF_2/LH_2. Performance studies showed no payload advantage for the LF_2/LH_2 system. It would appear then that, if the first-stage system is LO_2/LH_2, little justification can be shown for the development of LF_2/LH_2 for second-stage application.

LATERAL STAGE

Lateral staging can be accomplished by different combinations of dropping either the engines or the propellant tanks of a cluster and using all engines during first-stage boost. By starting all engines on the ground and using at least part of them during the second stage, engine reliability probably can be improved. This factor will be included as part of the analysis which determines the cost per pound of payload in orbit.

After reviewing studies conducted by Douglas on methods of booster mass fraction improvement by lateral staging, an approach utilizing systems simplicity was decided upon. The early booster mass fraction improvement studies were based upon the J-2 engine; however, the final report on this study placed greater emphasis on the plug nozzle type of engine. This difference does not greatly affect the fundamental comparison of the types of lateral stages. The discussion of the systems associated with one generally will also apply to the other. The four-around-one arrangement shown in C of Figure 50, which used two engines on each of the four modules and a single J-2 engine on the center tank, was rejected because of the reduced performance due to extremely low thrusts which were necessary in the second stage. After this investigation, the work was centered about a three-module vehicle with three engines on each module. To avoid interconnecting the tanks, with the valves and disconnects which reduce reliability, the center tank has been increased in size so that its engines burn during first stage as well as second stage. The payload weight used in the Douglas study is 70,000 pounds, while the first iteration on this configuration is based upon a payload weight of 77,000 pounds. Both concepts are semi-modular in that one employs common tanks while the other employs common engines.

Although no design or performance work has been carried out, it is possible that higher reliability could be achieved by substituting solids at the side position of a lateral stage. With the solid propellant system's shorter burn time, it is reasonable to expect that a higher performance mass ratio of the second stage would be required. Since propellant consumption rate is constant, less second-stage ullage space will exist when faster burning rates are used during first-stage boost.

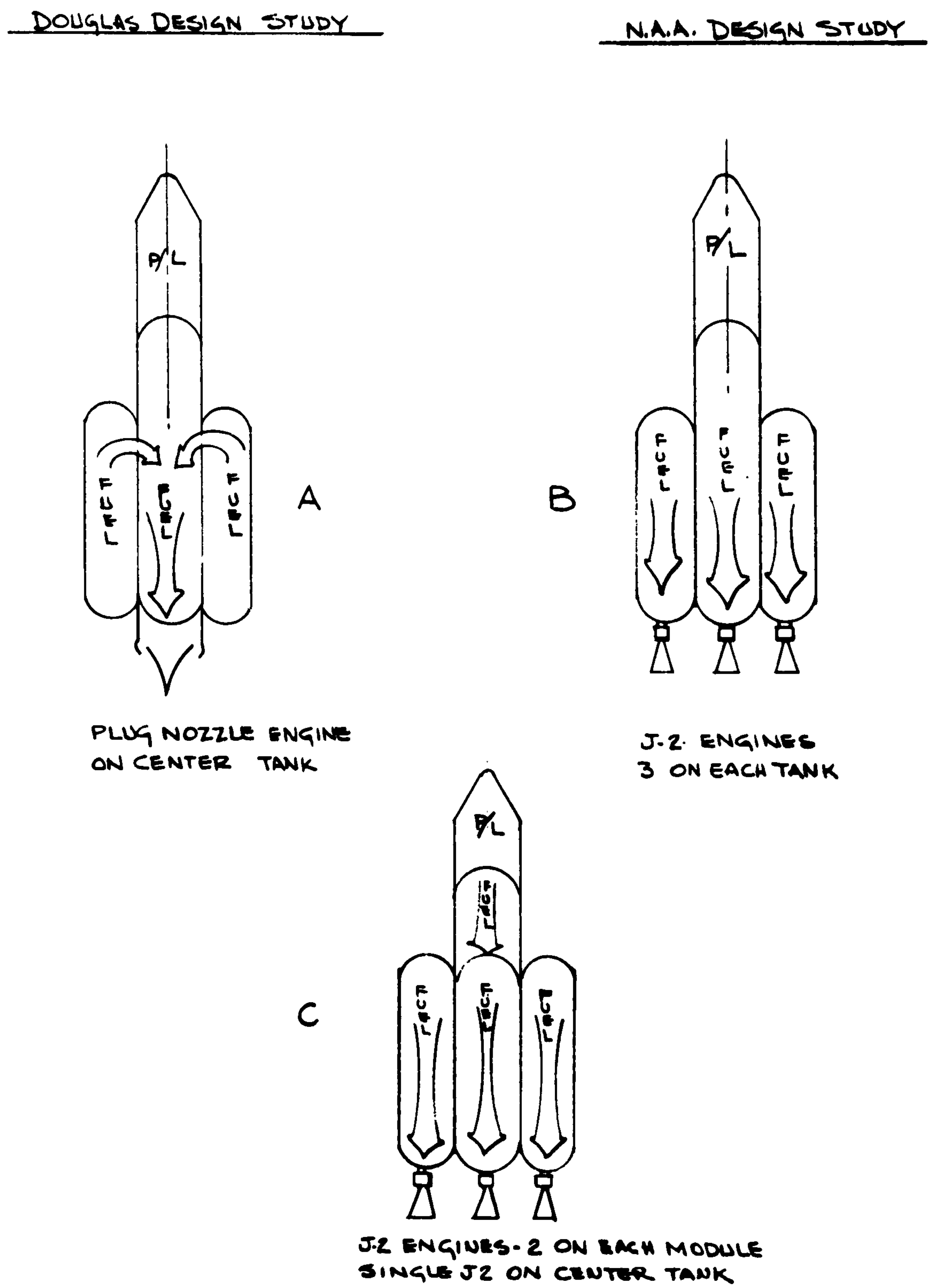


Figure 50. Lateral Stages

Small difference in detail are not the purpose of the comparison although some variations are included in engine selection and separation system. In all the comparisons, the propellant tanks are to be the bipropellant type, that is, each tank contains both oxidizer and fuel. In comparing one type of lateral stage with the conventional two-stage tandem arrangement, as described in the Douglas report, results showed that an increase in payload is achieved when lateral stages of the eight-around-one type are used. With the eight-around-one type, the tanks are dropped in pairs while all engines remain with the center tank structure. This amounts to a four-stage drop-off sequence which is more efficient in performance than the two-around-one arrangement. However, when evaluated from the more practical standpoint, which included considerations of vehicle drag and separation reliability, the number of drops was reduced. The final report in the Douglas study presented the performance of a two-around-one configuration where the center tank was fed from the outer tanks during first-stage boost. By using the weights for this configuration, which were supplied by Douglas, a direct comparison of performance can be obtained. The cost per pound in orbit can be obtained from the data in Table 25.

In the North American Aviation design the J-2 engines have been uprated to provide the required 166,000 pounds of thrust at sea level. This change is predicated on an increase in chamber pressure to provide the upgraded performance. To carry this change through the study and provide a more accurate comparison between the drop-tank version and the drop-module version, the J-2 engine weights have been increased to provide for the chamber-pressure increase.

At the time the North American Aviation lateral design was started, it was expected that this was the type of propulsion system which would be used in the final Douglas design. Early progress reports indicated this to be the case. However, the final report was based upon a design which mounted a single-plug nozzle engine on the center stage as illustrated in A of Figure 50.

In the case of the dropped tank version, the plug nozzle derives some benefits in specific impulse due to its flexible high-and low-altitude performance. However, end boost T/W will be high unless throttling is achieved. It also lends itself to an efficient method of support on the center tank. In the dropped-module version, it was not possible to use a plug nozzle wherein segments could be dropped as part of each outer tank. Although it is possible to use a plug nozzle on each tank, the interaction between the engines would appear to preclude such a concept.

At the time the dropped tank version was being designed by Douglas, studies were actively under way from which cost and performance data could be readily obtained. Therefore, the J-2 engine, with its bell nozzle, was selected for comparison purposes. Since any plugged engine would not be available until a later time period, it was justifiable to upgrade the performance of the J-2 engine on the assumption that it would be available concurrently with the plug engine.

To develop the cost comparison on lateral stages, an approach slightly different from that used in the Douglas study is shown by the three-tank arrangement in Figure A-12. To avoid any systems interconnection between the three tanks, B of Figure 50 shows each tank supporting its own engine structurally as well as from the system standpoint. Three gimbaled J-2 engines have been placed on each of the three tanks resulting in a total thrust of 1.5 million pounds. During first-stage boost, all nine engines are operating, after which time the outer two tanks with their engines are forced laterally away from the center tank.

Separation

To avoid the possible problems of binding and friction associated with mechanical means of translation, a pneumatic device has been proposed. After the fuel has been consumed in the two side tanks, the separation will consist essentially of two modes. The first is severing of the tank attachment structure by a shape charge. The second is the inflation of a mylar bag within the intertank structure. To provide rigidity for this box-like structure, the intercostals are made in two pieces to transmit shear but provide no resistance to separation. This is accomplished by providing bearing surfaces in a saw-tooth fashion as shown in the rear view of the drawing. It is important when using primacord or shape charges that the portion of the material to be severed be of a constant thickness. This can be accomplished by providing the two-piece intercostals.

Systems

Since no primary systems are ever required to cross between the center tank and the outer tanks, it is not necessary to develop any means of preventing flow of propellant or pressurizing gas from one tank to the other. It is possible that some secondary systems will require a certain amount of interconnection between the tanks. Generally, the line sizes involved for these systems are small and pull-away connectors with built-in check valves can be used. Each of the three tanks carries its own liquid helium system for pressurizing the LO_2 tank while heated LH_2 is used for pressurizing the LH_2 tank.

Structure

All tanks are fabricated of a semimoncoque aluminum in the cylindrical portions while the divider portion is made of welded aluminum honeycomb. To transmit the axial loads between tanks, the thrust is resolved into uniformly distributed shear forces along the length of the outer tank. This shear is transmitted to the aforementioned box structure, which can be attached to the adjoining tank with mechanical connectors at the launch site.

Engines

The J-2 engine used for comparison purposes on this configuration is considered to have a thrust of 166,000 pounds at sea level. This thrust can be achieved only if an expansion ratio of 8:1 in the nozzle is used. The engine, as shown in Figure A-12, presents an expansion ratio of 27:1. This position is justified by assuming that the chamber pressure in the engine will be up-graded to a point where the 166,000 pounds of thrust can be achieved. This represents an increase of 200 psi (the present chamber pressure is 600 psi; the required future pressure would be 800 psi).

The heat required by the LH_2 to pressurize the LH_2 tank is obtained from the unit which is shown as part of the J-2 engine. This type of pressurization system has been used for all configurations in this study which use the J-2 engine.

SINGLE STAGE

The greatest utilization of propellant specific impulse can be gained with the high-pressure engine by operating it at its maximum expansion ratio. This has been done in the single-stage-to-orbit configuration where a four barreled engine is directed into the secondary expansion nozzle, which is part of the vehicle airframe. In order to improve the sea level thrust characteristics, this nozzle has been designed with a perforated air bleed system. By using this device, separation development at low altitude will not occur, thus keeping the thrust at its maximum.

The propellant tanks on the configuration shown in Figure A-2 have been sized based upon a mixture ratio of 7:1. It had been anticipated that by increasing the propellant bulk density, sufficient improvements in booster mass ratio would be achieved so that the loss in propellant Isp could be compensated for. As a result of these trade-off studies, it was found that the booster mass ratio did not increase fast enough to compensate for the reduced Isp associated with this mixture ratio. Consequently, a net payload decrement of 500 pounds was incurred in the design which used a mixture ratio of 7:1. This difference is actually small compared to the total payload

weight of 20,000 to 30,000 pounds; however, it does indicate that specific impulse is the most important influencing factor. Since the payload was reduced by 500 pounds when the 7:1 mixture ratio was used, all subsequent performance work was based upon a mixture ratio of 5:1.

The control system used to establish the base point for costing the single-stage-to-orbit vehicle was assumed to be similar to the system used for the tandem designs. Many differences come into play in the single-stage vehicle which can greatly affect the detail definition of the control system but will not have such an effect upon the cost optimizing portion of the study. It is in the area of reliability of the control system where the greatest effects upon the cost optimization study will be felt, and future detailed definition of various control systems for use in this application are required to establish this.

In addition to correcting for the wind shears encountered during boost, the control system must also cancel the adverse effects of thrust misalignment and fuel sloshing. It must also provide the forces to turn the vehicle on its trajectory to orbit.

The control system used on the single-stage-to-orbit vehicle is especially important. Considering the sensitivity of vehicle performance to the booster mass ratio, it is obvious that payload definition is required. It is not possible to use the same control system generalizations on a single-stage vehicle as those used for tandem multistaged designs, where the system can be tailored to the requirements of each stage. Therefore, the high correcting moment requirements which are set by conditions at low altitudes become a part of the control system, which is carried for the total boost time of 270 seconds.

In any vehicle, whether single or multiple staged, the use of gimbaled engines provides an excellent source of control moment for pitch and yaw attitude corrections. On many designs the exhaust gases from the turbine are used to provide the necessary roll control forces. By these methods, a virtually unlimited force vector can be provided with a minimum of added systems. The greatest savings that such a system will contribute need not be accurately established. When the same system and propellant which feeds the main rocket engines are used from this source the control requirements can be estimated by degrading the total impulse available for vehicle acceleration. The systems and actuators necessary to move the engine are also considered in this process.

To achieve the greatest benefits from the high-pressure engine in a single-stage-to-orbit application, it is necessary to increase the engine expansion ratio by the addition of a nozzle extension. The nozzle extension

shown in Figure A-2 is part of the fairing around the four-barrel engine. By using this type of installation, the secondary nozzle becomes a part of the vehicle airframe which, because of its large diameter, is better adapted to carrying this nozzle.

In addition to the differences in propellant feed system, which are required with the high-pressure engines if they are to be gimbaled, this nozzle extension imposes another restriction on engine gimbaling. Therefore, on the basic configuration the use of auxiliary hyrodgen peroxide nozzles was proposed as shown in A of Figure A-3. When the nozzles are located near the aft part of the vehicle, the moment arm available for pitch and yaw correcting forces is only 335 inches. By using an independent vector control system at the forward end of the vehicle, an effective moment arm of 830 inches can be achieved at the time of maximum wind shear.

The possibility of integrating the pitch and yaw control system into the payload is shown in configuration C of Figure A-3. In this way, the effective moment arm is increased while the payload is attached to the booster and the same control system can be used while the payload is being injected into orbit.

Low-density vehicles, such as the one which uses LO_2/LH_2 as a propellant, are more sensitive to wind shears and gusts and consequently will require greater forces for corrective control than a higher density vehicle such as LO_2/RP.

Maximum wind shears occur at altitudes of approximately 35,000 to 40,000 feet. The single-stage-to-orbit can be expected to be similar to a two-stage vehicle at this point in its trajectory; therefore, multistage vehicle dynamics can be used as a basis for comparison of various methods of attitude control.

Consideration was given to increasing the fineness ratio of the vehicle in an attempt to increase the static stability and reduce the requirements of the control system. This design is shown in configuration B of Figure A-3. The large exit area of the nozzle, with an expansion ratio of 400:1, can be used to help shift the aerodynamic center rearward and thus improve the stability. The arrangement was considered undesirable because of the low volumetric efficiency associated with propellant tanks with a high-fineness ratio.

ENGINE INSTALLATION

One of the problems encountered in evaluating the high-pressure Pratt & Whitney engine was the method whereby this engine could be gimbaled without degrading the reliability of the main propellant feed lines. Since the engine is designed with the intake in the forward end, it was necessary to show methods where the flexing of these intake lines could be held to a minimum during gimbaling conditions. In Figure 51, the feed line has been routed near the plane of the engine gimbal points. In so doing, the deflection at the inlet lines has been reduced to a point where high reliability can be achieved because these lines are operating through the angle for which they were originally designed. If the feed lines were routed directly from the front of the engine to the tank, movements as high as 2 inches would have to be taken out by the line with consequent reduction in reliability. In attempting to keep this line routing as simple as possible, some designs were evolved which included omega (Ω) bends, but these were rejected due to the high torque imposed upon the flex connection. This method of line routing will cause more losses due to unused propellant in the lines. However, there are no known flex lines which could be attached at the forward end of the engine which would accept the required 4 degree gimbal angle. Figure 52 shows that some benefits can be derived from the standpoint of engine actuation by using the center pivot point type of engine mount. With this method, the actuators can be placed in the protected region inside the engine mount cone. The actuators then can be attached to the upward projecting portion of the engine and thus take advantage of the long moment arm. When comparing this type of engine installation with that used on the J-2 engine, it should be noted that a dual gimbal ring type of mount could also be designed for the J-2. Therefore, any advantage in reduced actuation forces are not necessarily associated with the high-pressure engine. The moment of inertia is reduced by pivoting the engine near its center of gravity. This polar moment of inertia probably would be less for the J-2 engine than for the high-pressure Pratt & Whitney engine with similar type mounts. Once this type of change had been made in an advanced version of the J-2 engine, it could claim the same component mounting advantages which have been discussed for the high-pressure engine. It no longer would be necessary to place a separate flame shield behind the engine mount within which the components are to be mounted. If the conical engine mount was used, it could be considered a flame shield for both the J-2 and the high-pressure engine.

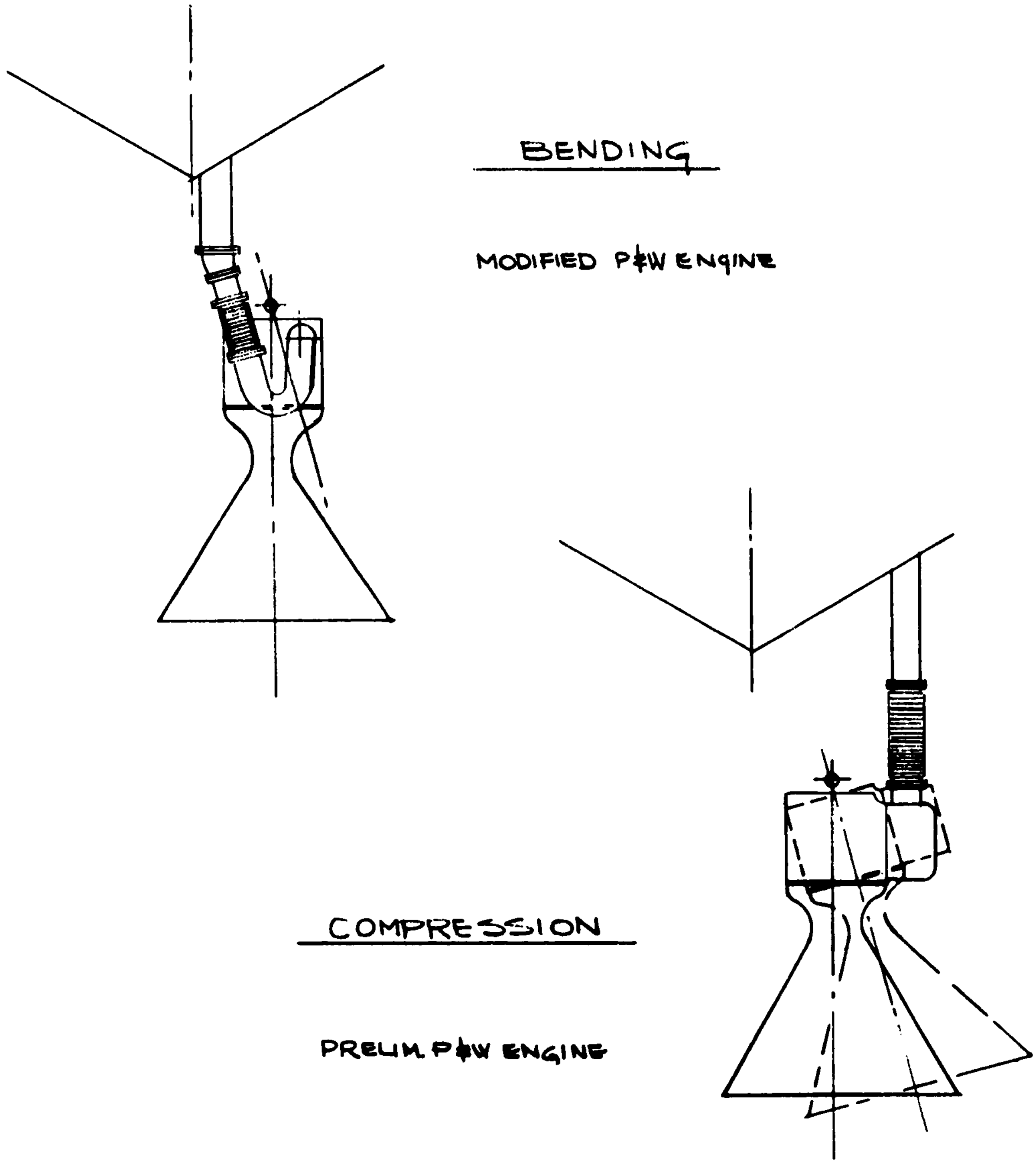


Figure 51. Line Flexing - Pratt & Whitney High-Pressure Engine

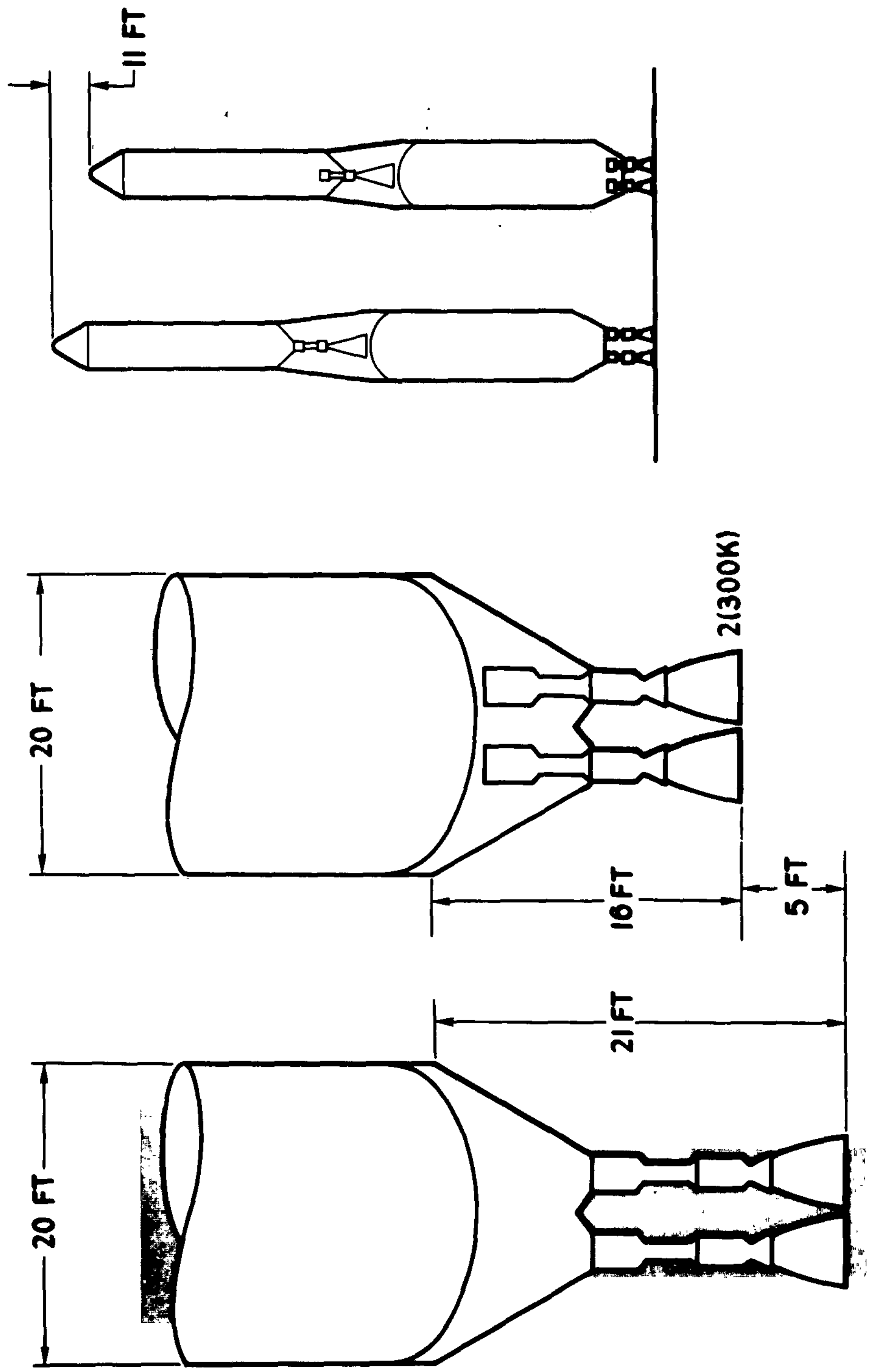


Figure 52. Comparison Chart - Engine Gimbaling for 600K Pratt & Whitney Engines

On the engine thrust structure shown in Figure A-1, two methods of gimbaling the Pratt & Whitney engine were studied.

The first method places a gimbal box similar to that used on the J-2 at the forward end of the engine. It has the advantage of producing less propellant line length and less flexure than the second method studied, which places a gimbal point at the forward end of the thrust chamber, which is a point approximately one-third of the distance from the forward end of the engine to the nozzle exit. This method results in longer propellant lines and slightly more line flexure than the first method, but it offers the following advantages:

1. Excursion of the nozzle exit for a given deflection angle is one-third less than for the first method.

2. The distance from the aft end of the LO_2 tank to the nozzle exit is about 5 feet less for a 300K engine, or an overall length of 11 feet in a two-stage system.

3. The method of gimbaling at the forward end will produce a high-bending moment in the engine structure between the primary and secondary thrust chambers. This will not occur if the engine is gimbaled near the center.

4. The extension of the forward portion of the engine into the thrust structure, as in the second method, produces an ideal location for gimbaling actuators and permits a wide selection of moment arms for the actuator attach point. In contrast, the first method limits the actuator to a very short moment arm.

600K PRATT & WHITNEY HIGH-PRESSURE ENGINE SYSTEM

The various layouts presented in the appendixes of this report show that a high-pressure engine utilizing the preburner concept presents some installation and control problems. The present method of controlling the thrust vector of an engine is to gimbal the engine or the nozzle. With the Pratt & Whitney High-Pressure engine, the gimbal point presents a problem. Figure A-1 depicts stacking of components, resulting in the least performance and weight degradation. However, the rotating machinery housings will not accept the loads imposed during gimbaling without excessive deflection. A location that appears feasible to accept these loads is in the area of the injector. This area, however, is located near the midpoint of the engine. Figure A-1 illustrates one possible method of gimbaling at this point. Another problem with this method is the deflection of the propellant lines.

An unacceptable amount of deflection is encountered unless the flex joints are located near the gimbal point. Such a location would result in increased system weight and line loss. Discussion with Pratt & Whitney indicates relocation of components is entirely feasible and that Figure A-1 represents the best engine installation from the manufacturer's point of view. The preliminary nature of the engine concept did not allow detail analysis and design of a gimbal system.

Various other methods of vehicle attitude control also were investigated, such as jet vanes, fluid injection, small attitude engine system, and rotatable solid motors. However, for the purpose of this study, it was assumed that the high-pressure engine was readily adaptable to a gimbal system. Figure 3 illustrates a comparison of 600K systems that were investigated in this study. One of the most interesting configurations was the single-stage-to-orbit case. The performance of this vehicle was based on the assumption that an engine system utilizing secondary expansion could be developed. This system consists of several high-pressure engine modules, clustered and incorporating multiple pumps or a single pump for all modules. The expansion ratio of the modules would be 23. A secondary expansion surface is provided in the form of a large single bell nozzle and an assumed area ratio of 400. If ambient pressure can be maintained in the center of the nozzle, optimum expansion will occur until the design area ratio is reached.

The high-pressure engine cycle is estimated to require pressures at the pump inlets that cannot be supplied by present state-of-the-art pressurization systems without undue weight penalties. One solution is either tank- or engine-mounted boost pumps. The exact performance degradation associated with this system has not been determined as the pump NPSH requirements are not established. Theoretical performance estimates were reduced a nominal amount to account for combustion efficiency, propellant bleed for pressurization, and power extraction for the boost inducer system.

CONCLUSIONS

By using a spherical engine mount (or gimbal rings), and supporting the engine near its center of gravity, it is possible to mount the high-pressure engine efficiently on single or multiple conical engine mounts. The weight saving is accomplished by using the engine mount cone to protect components which are heat or flame sensitive. Provision of a separate flame shield for these components therefore will not be required.

These methods of gimbaling the high-pressure engines will require the installation of flex lines which are different from those presently in use, but which appear to be feasible and reliable.

By selecting simple propellant feed systems, there are noticeable economic gains possible with lateral stages. Problems which remain are thermal strains in the tank material due to different time of emptying. Attitude control requirements are different in one plane than another. Solid propellants lend themselves to use in the side position of such boost systems.

Selecting a duty cycle on the single-stage-to-orbit vehicle is important in weights analysis.

Solids for use in the first stage of a tandem booster present no major problem. The stability and control problem is increased, however, because of the low-density propellant in the second stage. This increases the length of the vehicle interstage and spreads the two dense masses (payload and solid first stage).

STRUCTURES

SUMMARY

The purpose of the structural analysis is to provide a sound basis for an accurate determination of the structural weight and costs involved in the various configurations under study. Analysis was made in sufficient detail to establish the validity of basic load paths and structural components.

Because of the vast number of configurations under study, a machine program has been utilized to calculate the weight and cost of all the basic structural components. However, hand-calculated basic points are required as machine inputs. In the initial phase of the study, base points established in the previous Edwards Air Force Base Advanced Propulsion System Study were employed. Booster tankage was of skin-stringer construction with ullage pressures of 35 psia and 50 psia respectively for the LH_2 and LO_2 tanks. In the later stages of this study, specific configurations were used for detail stress analysis and subsequent use as base points for the machine program to determine its influence on Systems Cost.

Here the booster tankage is of sandwich construction and tank ullage pressures are reduced to the minimum needed to meet NPSH requirements of the pumps. The new ullage pressures are approximately 30 psia in both LH_2 and LO_2 tanks for a typical arrangement. Figure 53 illustrates the weight saving by use of sandwich construction instead of skin-stringer. Figure 54 indicates the weight reduction by use of lower-ullage pressures. The overall effects per pound of payload is shown by bar charts in Figure 65 in the weights section of this report.

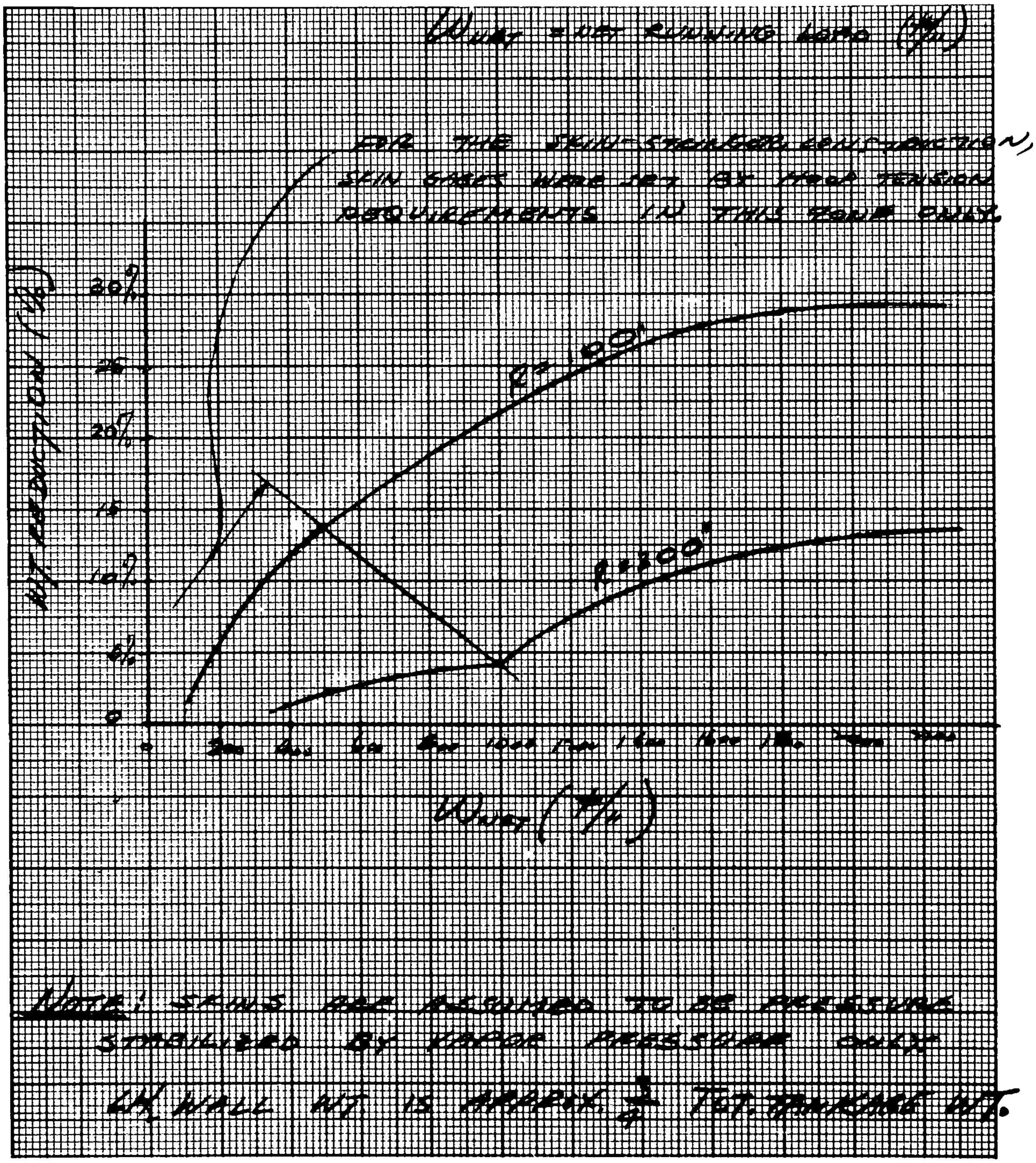


Figure 53. Liquid Hydrogen Wall Weight Reduction: Sandwich Versus Skin-Stringer

[redacted]

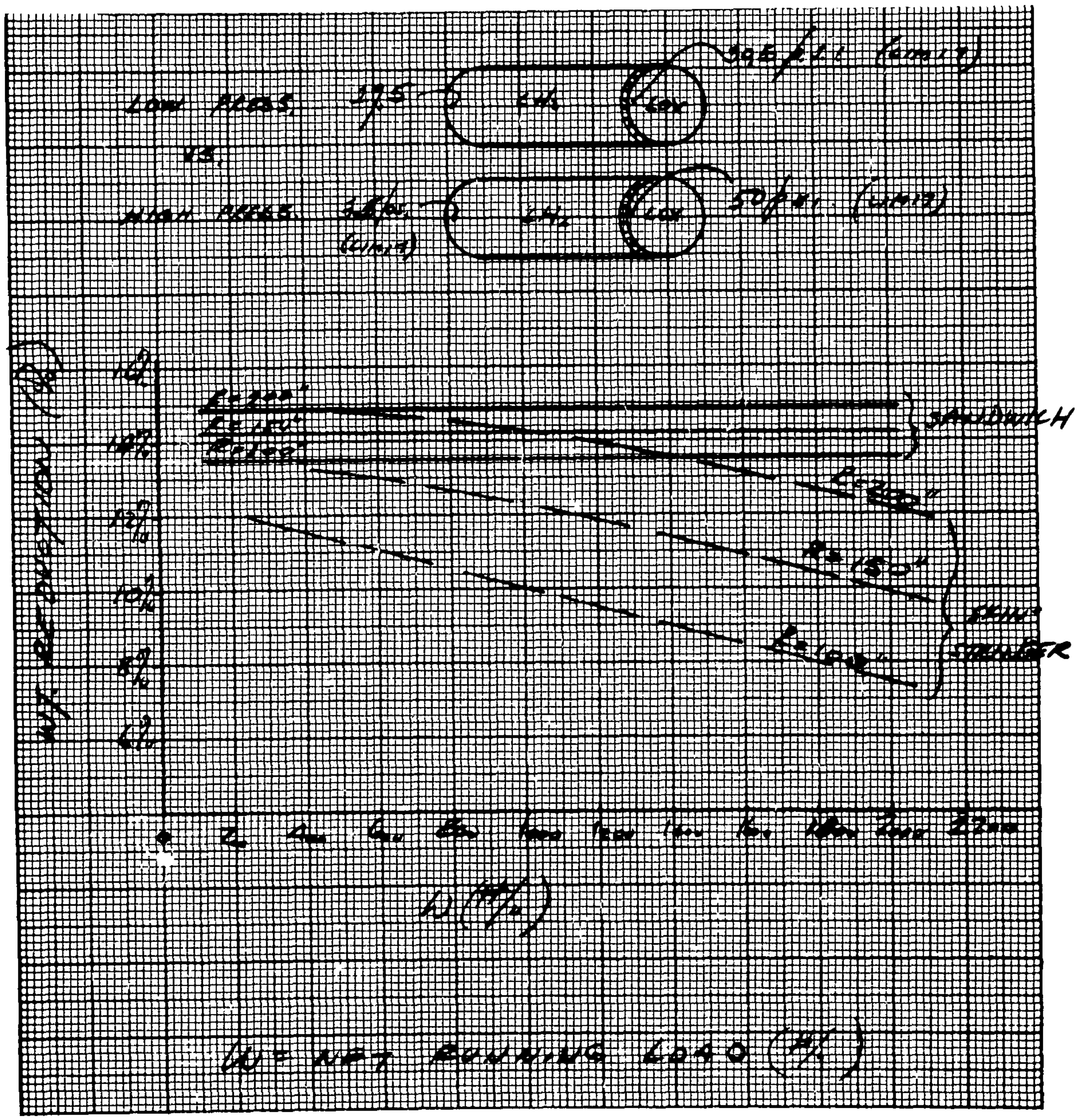


Figure 54. Liquid Hydrogen Liquid Oxygen Tankage Construction Comparison

[redacted]

DESIGN CRITERIA

FACTOR OF SAFETY

Safety factors of 1.10 and 1.35 have been applied to limit load for yield and ultimate load respectively. Allowables have been based on guaranteed minimum properties at the temperatures indicated in Table 12.

Table 12. Tank Wall Temperature in Degrees

Stage	Condition	RP-1	LOX	LH_2	Thrust Structure	Interstage (Skin-stringer) Structure
I	End-boost I	300	0	-150	150	300
II	End-boost II	150	-150	-300	150	300

Sufficient weight has been allowed for insulation of LH_2 tanks and flame shielding of the engines.

MATERIALS

The choice of material was based on the attainment of a minimum weight design consistent with producing an economical, high-confidence structure with a short development time. Aluminum alloy 2014-T6 was selected as the basic material. This alloy combines the desirable high-strength-to-weight properties at 300 degrees and at cryogenic temperatures in a material that is easily machined, welded, or extruded.

LOADING CONDITIONS

The following loading conditions were investigated:

1. Prelaunch, i.e., free-standing, unpressurized, and subjected to a 40 knot wind with a 20 knot gust.

2. Mid-boost stage I

3. End-boost stage I

4. Begin- and end-boost stage II. In general, the critical loading condition occurs under the maximum acceleration of end-boost and is predominantly axial.

STRUCTURAL COMPONENTS

TANK WALLS

The tank walls are designed for free-standing while unpressurized during the prelaunch phase and for body bending and axial loads encountered during flight. A monocoque-sandwich construction was found to have a small weight advantage over conventional skin-stringer construction for the tank walls as illustrated in Figure 53. The skin gages were set by hoop tension requirements resulting from internal pressure with sufficient core material being added to stabilize the facing sheets under axial compressive loads.

HEADERS AND DIVIDERS

A typical tank arrangement is illustrated in Figure 55. Only the forward and aft headers are subjected to bursting pressure and are of membrane construction with an elliptical shape to minimize weight. In configurations where a portion of the aft header is incorporated as part of the thrust structure and subjected to compressive loads, these segments are of sandwich construction.

The aft header is generally critical at end boost when it is subjected to ullage pressure plus dynamic head. The center divider is subjected to both bursting and collapsing pressures, and an ellipsoidal sandwich construction produced the minimum weight at this point. The facing sheet gages were determined as the minimum required to withstand the tensile stresses resulting from the bursting pressure, while core thickness was minimized to that required to stabilize the facing sheets under the compression stresses induced by the collapsing pressure.

THRUST STRUCTURE

The function of the thrust structure is to transmit the concentrated thrust of the engines to the booster outer shell, distributing these loads as uniformly and as directly as possible with a minimum-weight structure.

For a single engine located at the tank centerline, a conical thrust structure is ideally suited (see Figure 56). Configuration A of Figure 56 illustrates a fitting at the apex which immediately achieves a uniform circumferential distribution and a frame at the tank wall which realigns the thrust to a longitudinal direction. This imposes a uniform radial outward load on the frame (see B of Figure 56). If a sandwich type construction is

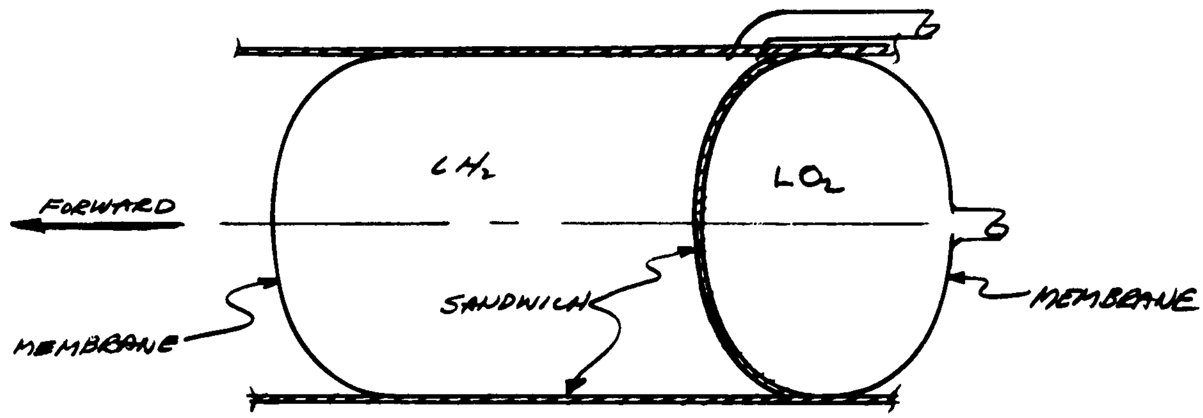


Figure 55. Typical Tank Arrangement

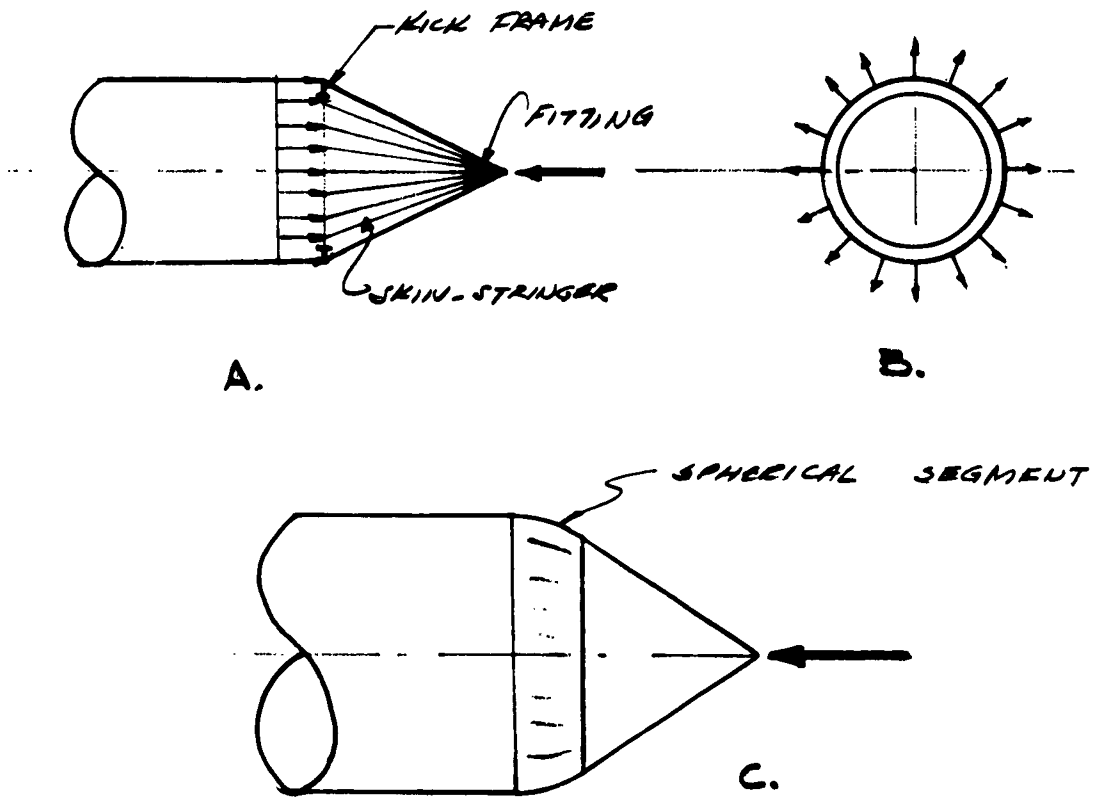


Figure 56. Single-Engine Thrust Structure

used, this frame may be eliminated by replacing the forward portion of the cone with a spherical segment (see C of Figure 56).

A multiengine cluster (four or more) requires the following elements: (1) tapered integral thrust longerons to accept the concentrated thrust at the engine pickup points; (2) a conical frustum (skin-stringer or sandwich) to shear load out of the thrust longerons and effect a uniform load distribution at the junction of the conical frustum and cylindrical tank (see A of Figure 57); and (3) two frames, one at each end of the conical segment, to realign the thrust loads as indicated in B and C of Figure 57.

The case of the two-engine tank presents special problems and has been handled in a unique manner. The single cone approach is obviously not possible, and the conical frustum approach of the multiengine system has two distinct disadvantages: (1) an excessively long conical frustum is required to achieve a uniform distribution at the cylinder from the two widely separated thrust longerons (see A of Figure 58); and (2) two frames are required, the aft frame being quite heavy due to the severity of the two-point loading conditions (see B of Figure 58).

The method devised to obviate these disadvantages was to use two intersecting cones with a large number of integral longerons. A fitting at the apex of each cone immediately achieves load diffusion and eliminates the need for a frame at the engine pickup points. A parabolic arch is required at the intersection of the two cones and an integral longeron is required in the cylindrical tank at its junction with the arch as illustrated in Figure 59. These, however, are fully utilized by locating the ground support attachments at the same joints. A uniformly loaded frame at the base of the dual cones completes the thrust structure.

INTERSTAGE STRUCTURE

The interstage structure is the most highly loaded portion of the booster shell. In addition to the usual body bending and axial loads, this structure is subjected to higher temperatures and receives no axial load reduction resulting from internal pressure. For tapered interstages, there is an additional increment of stringer load because of the taper as well as higher-skin temperatures resulting from the increased aerodynamic pressure. A skin-stringer construction is used here to minimize thermal degradation. Figure 60 illustrates external, integrally extruded stringers rivetted to internal frames.

TANK CLUSTERING

The purpose of tank clustering is to obtain a vehicle of high-boost capacity without the cost and time required for the development of a new

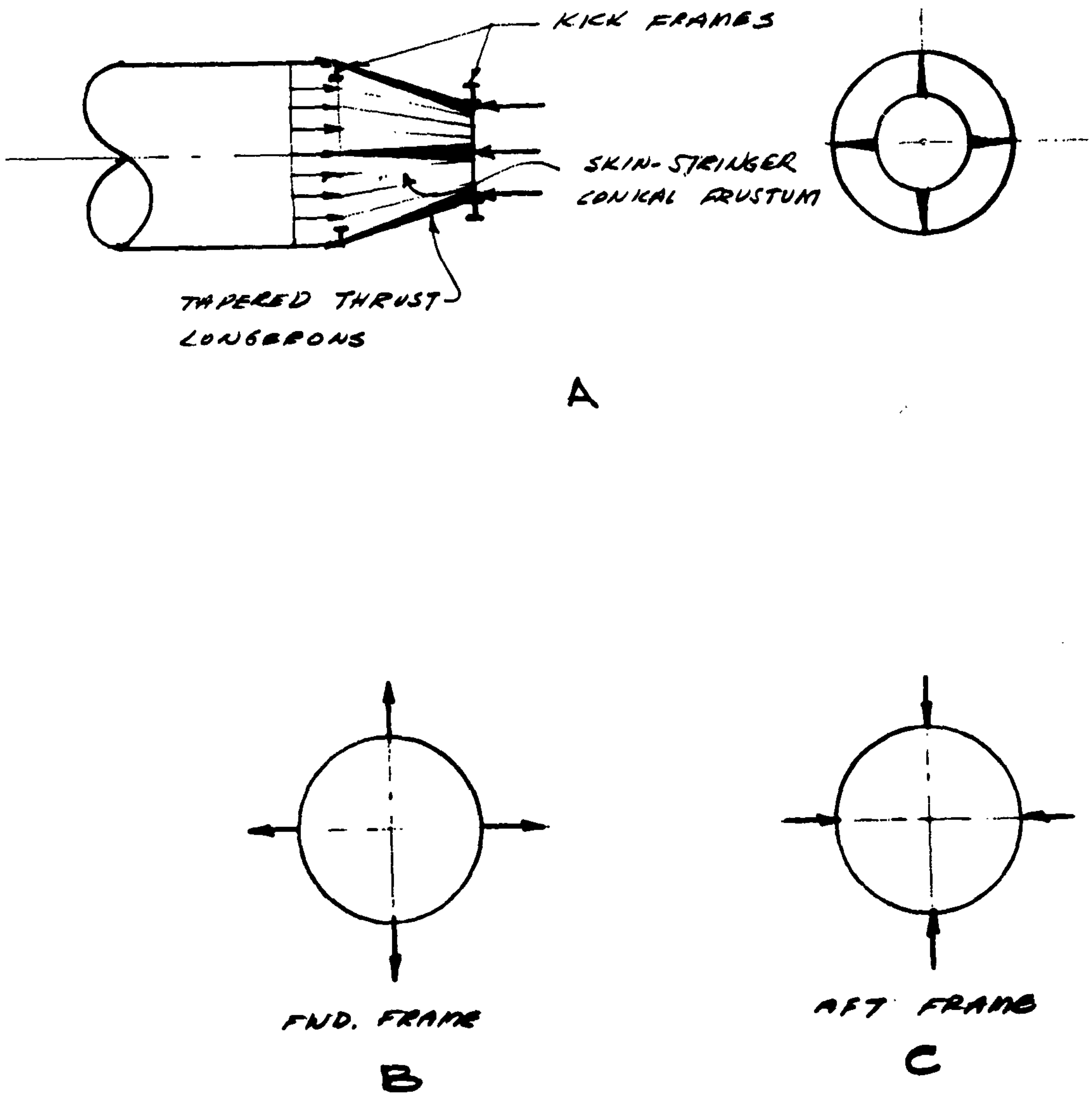


Figure 57. Multi-Engine Thrust Structure

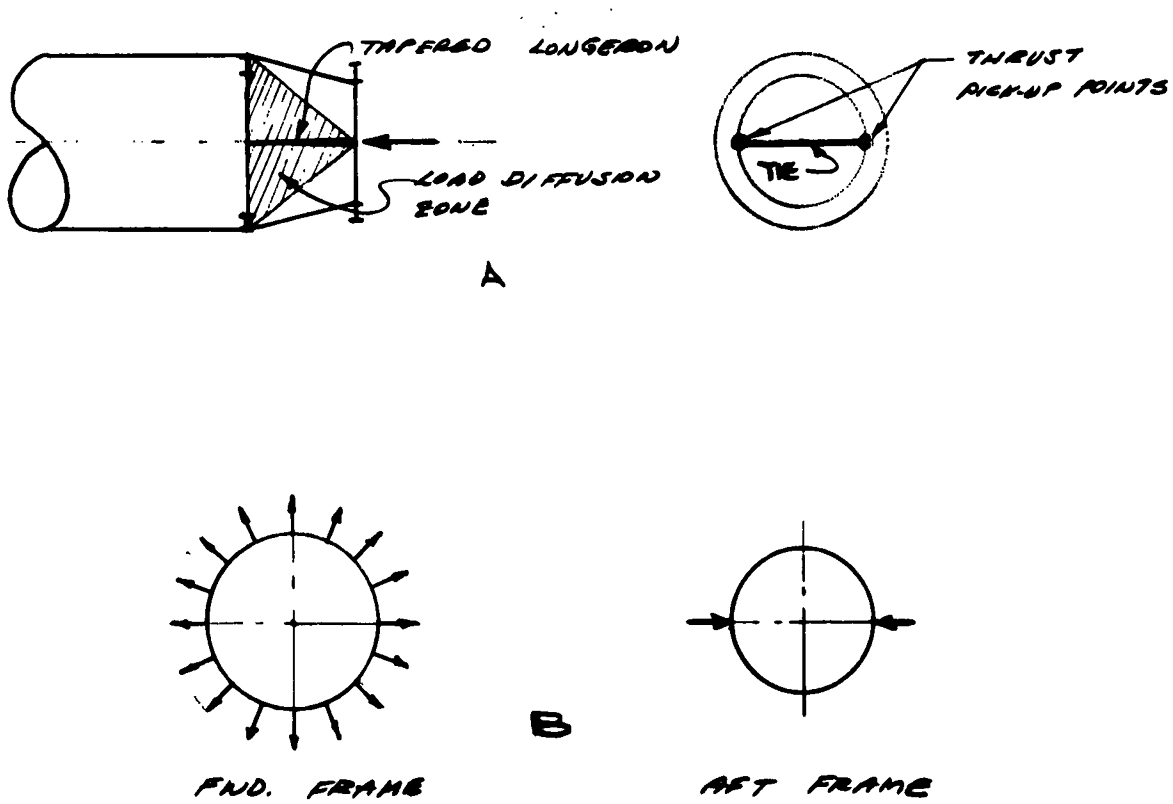


Figure 58. Two-Engine Thrust Structure

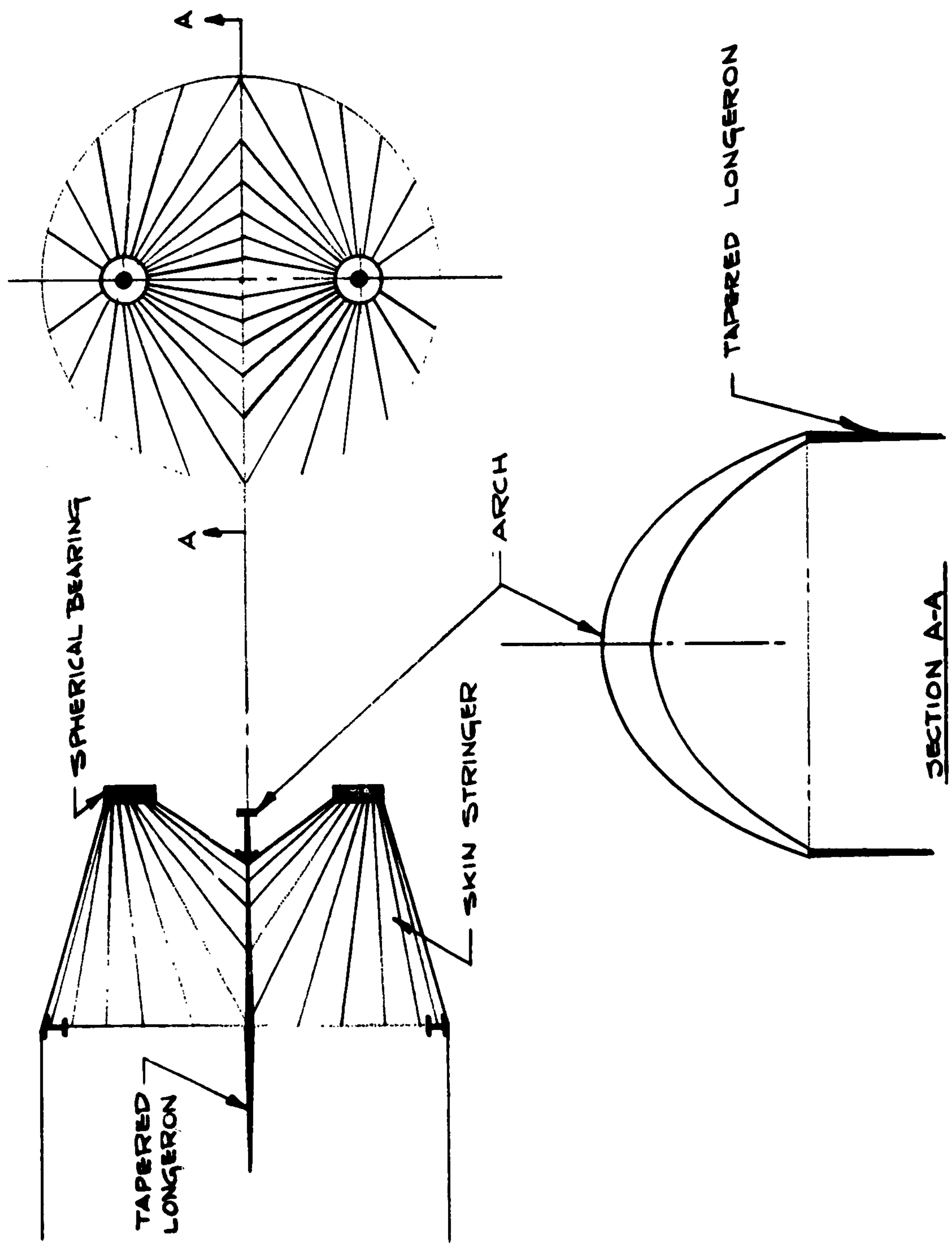


Figure 59. Dual Cone Thrust Structure

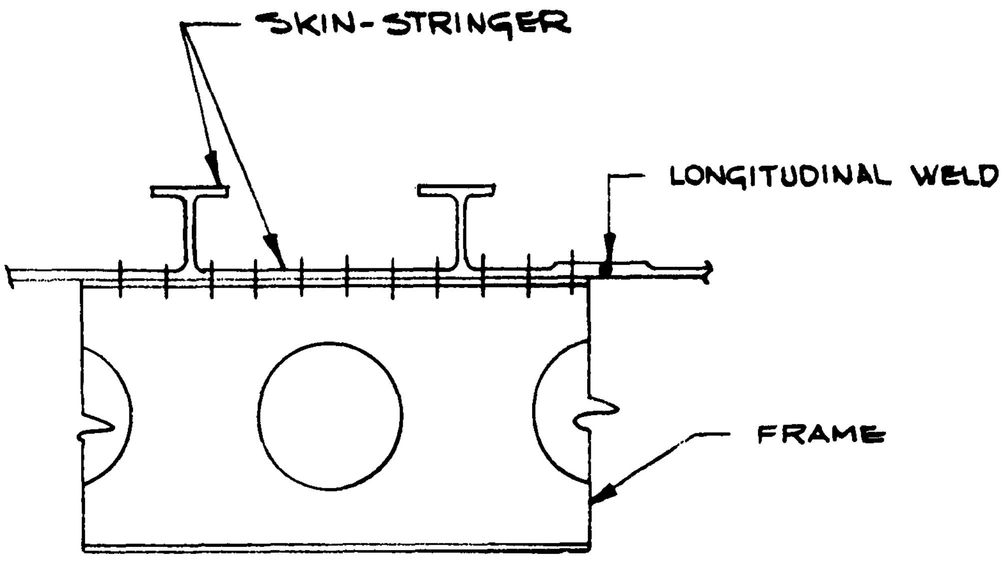


Figure 60. Typical Skin-Stringer Construction

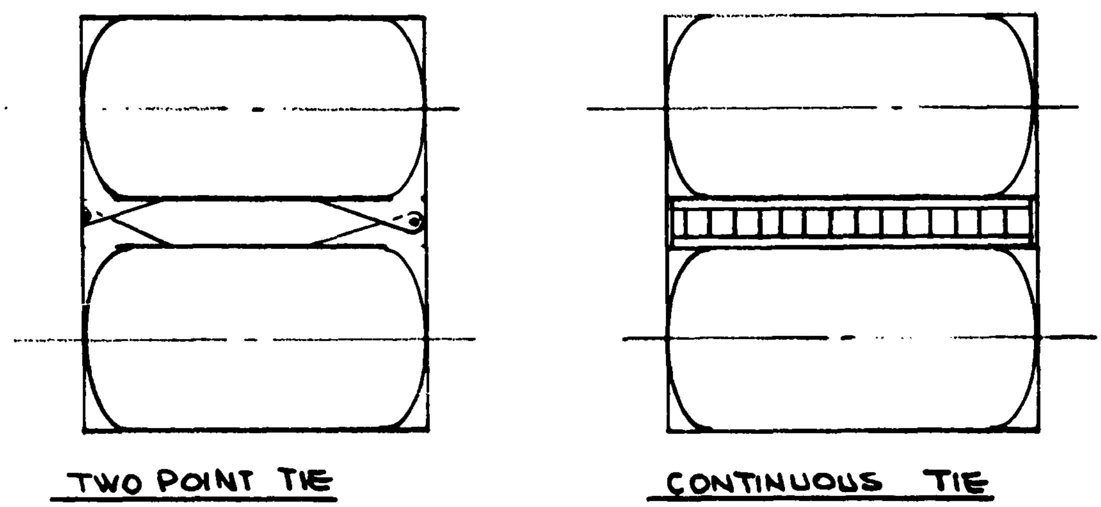


Figure 61. Tank Clustering Arrangement

engine and boost vehicle. In the modular concept, all tank units are identical resulting in a weight penalty in the first-stage tanks.

Two clustering methods were investigated, a concentrated shear tie between tanks at both ends of each tank and a full length continuous shear tie between tanks (see Figure 61). The continuous shear tie, while presenting a more difficult assembly operation, proved to be more desirable for several reasons. In the concentrated shear tie method, large loads are induced at the tie point in order to enforce matching deflections at adjoining tanks (see Figure 62). This necessitates the use of long tank longerons and stabilizing frames to distribute these loads, which are members not needed in the second-stage module. The use of the continuous tie approach provides a monolithic bending structure of 100 percent greater rigidity. The shear loads inducedare smaller and are uniformly distributed requiring no additional internal tank structure. By using two shear webs at each tie, an enclosed rigid torque box is created between each tank providing a continuous torsional restraint throughout each tank (see Figure 63).

The greatest problem with regard to clustering is the interstage area, and it is in this area that the major portion of the weight penalty occurs. Figure 64 illustrates an interstage for connecting a single tank to a cluster of four tanks of equal diameter.

The forward end of each of the tanks is surmounted by a conical section; all thrust and shear loads are concentrated in a fitting at the apex, which provides a convenient pickup point for general handling operations. These fittings are then attached to longerons which are integral to a conical frustum at the base of the upper stage which serves to distribute the load uniformly to the upper-stage tank wall. Frames at the forward and aft end of the conical frustum react the kick loads which result from the change in direction of the load path.

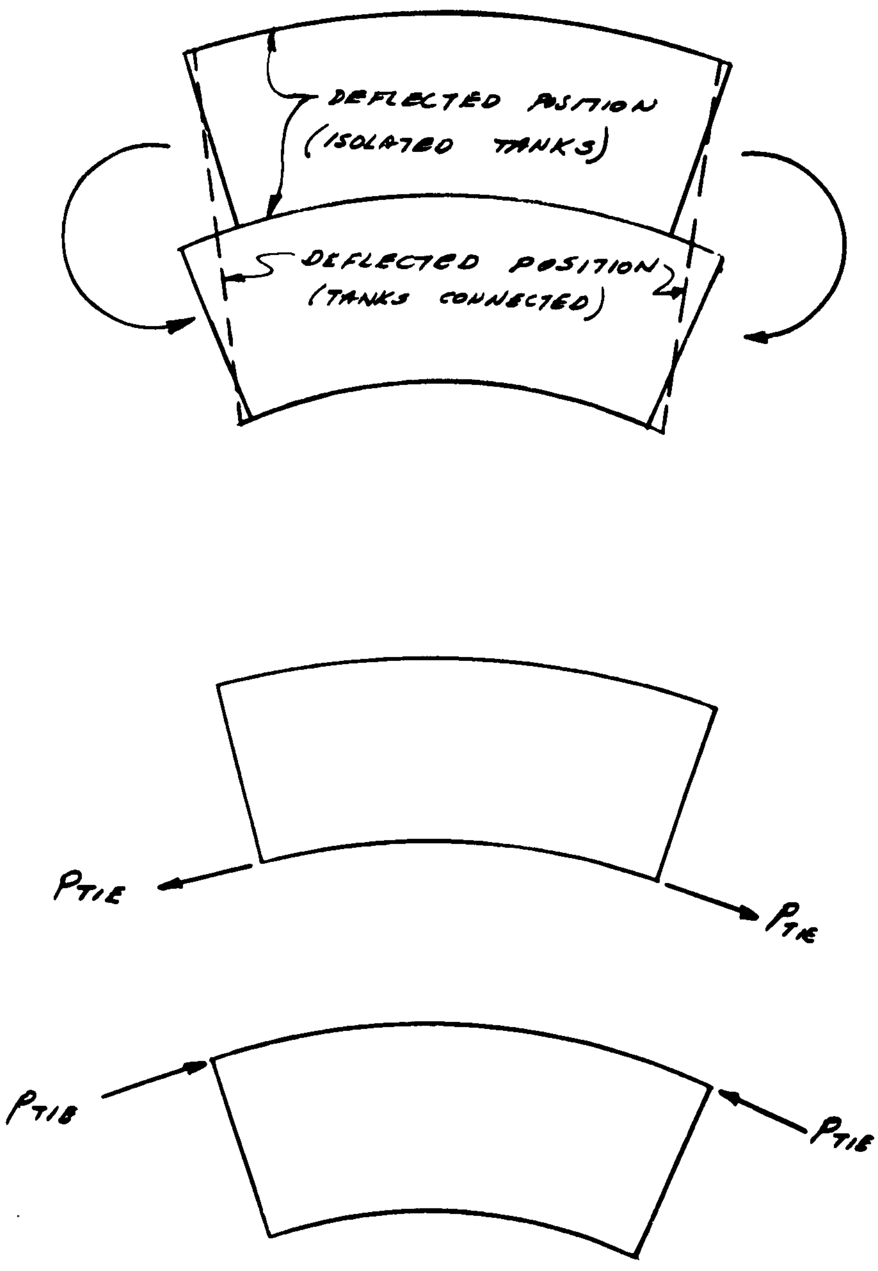


Figure 62. Two-Point Tie

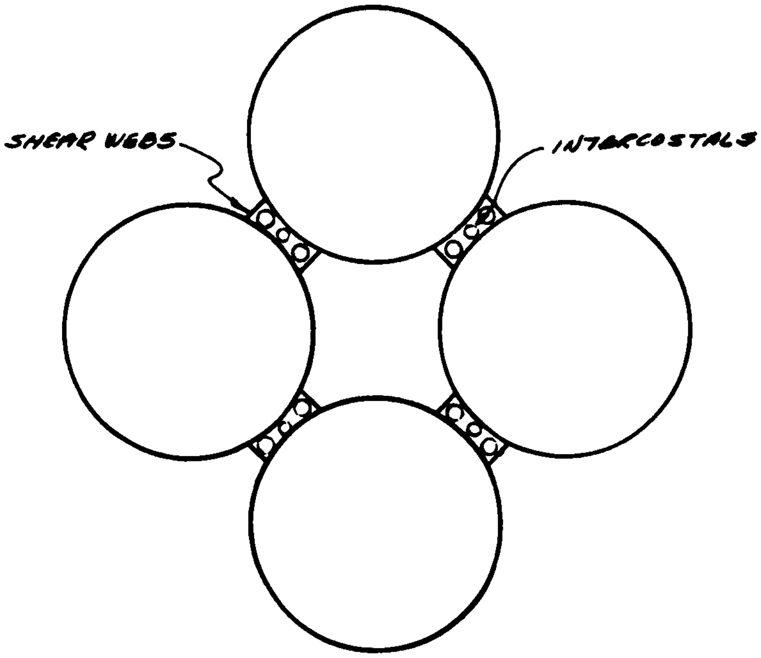


Figure 63. Clustered Tanks - Shear Webs

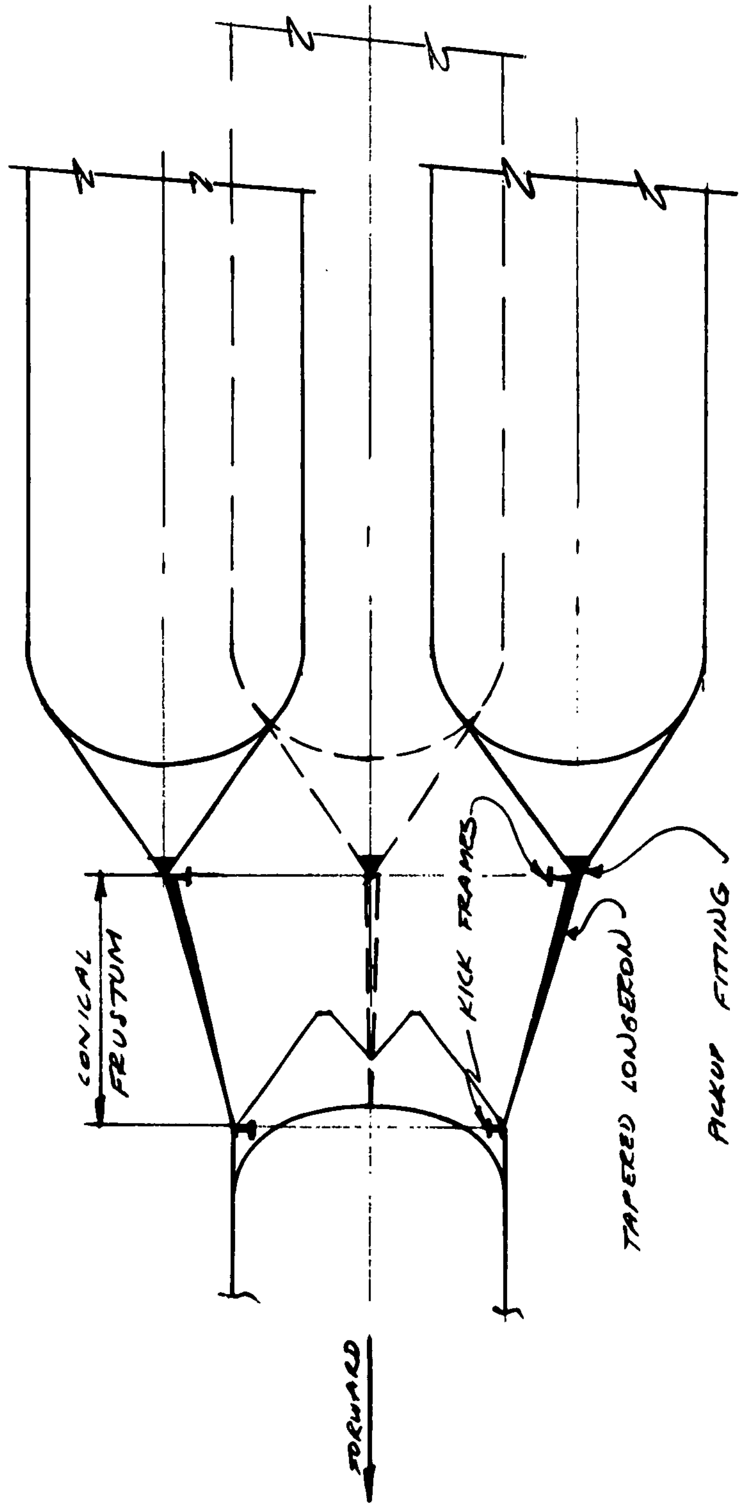


Figure 64. Clustered Tank Interstage

WEIGHTS

SUMMARY

The weight studies were conducted for single-tank boosters at two-basic thrust levels, 600 thousand and 1.5 million pounds. Alternate thrust levels of 800 thousand and 3 million pounds were examined as well as clustered modules at the 600 thousand-pound thrust level.

Several tank designs were analyzed for 600K and 1500K thrust levels. The various tank designs had a relatively small effect on the payload. Figure 65 graphically presents this effect on a few of 600K configurations. It was also determined that the larger tanks effected their respective payloads by a smaller percentage.

METHODOLOGY

The prime objective of the study was to establish the most efficient two-stage configuration for each of the two-basic thrust levels. A machine program was developed to compare the many engine configurations as well as the various upper-stage thrust levels for each booster concept as soon as possible. Propellant-mass fraction (ν_1, ν_2) were derived from estimated booster-mass fractions (ν_B) for each stage. A calculated weight was established for a typical vehicle of the 600K thrust level based on the design parameters specified in the structures section. The previous Edwards Air Force Base Study was used as a basis for the 1500K booster weights. The 1500K booster tanks were designed to the following criteria: skin-stringer construction, conical or sperical bulkheads, 50 psia pressure for the liquid-oxygen tank, and 35 psia pressure for the liquid-hydrogen tank. The payload effect of these design differences are shown in Figure 65.

All engine data are those of the vendors. Figure 66 illustrates the weight versus thrust of the Pratt & Whitney, 3000 psi chamber-pressure engine. This weight data was used in the analysis of the configurations employing this type engine.

MASS FRACTIONS

Booster-mass fraction V_b is defined as:

$$\frac{\text{Stage Propellant Weight}}{\text{Stage Gross Weight-Stage Payload Weight}}$$

The weight data used in the performance program analyses are in the form of booster-mass fraction curves for each configuration for variable amounts of propellant weight. Mass fraction curves for the first and second stages of various configurations in each thrust level are shown in Figures 67, 68, and 69. These mass fraction curves are based on machine-program weight data.

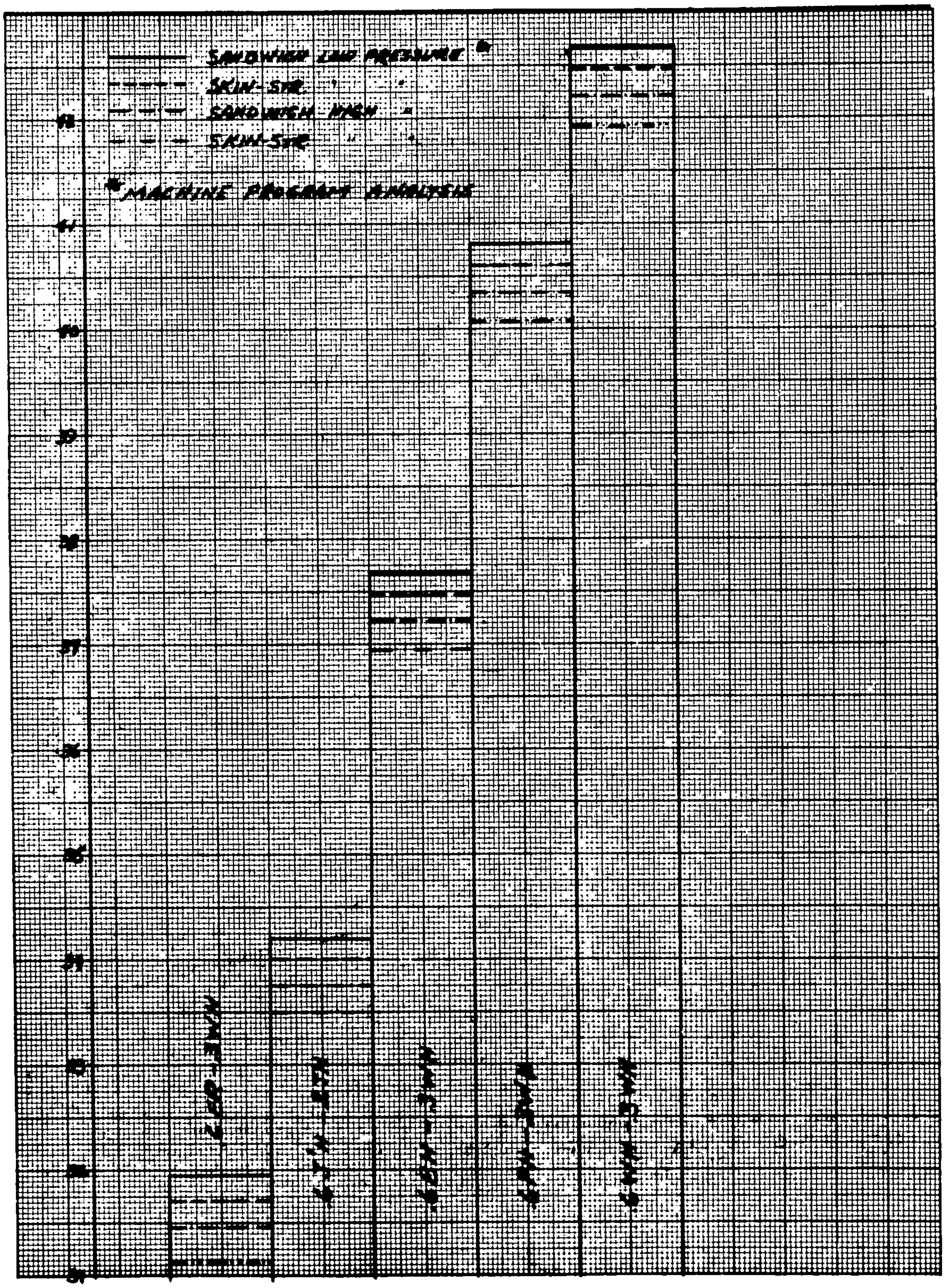


Figure 65. Payload Variations for Various Design Concepts

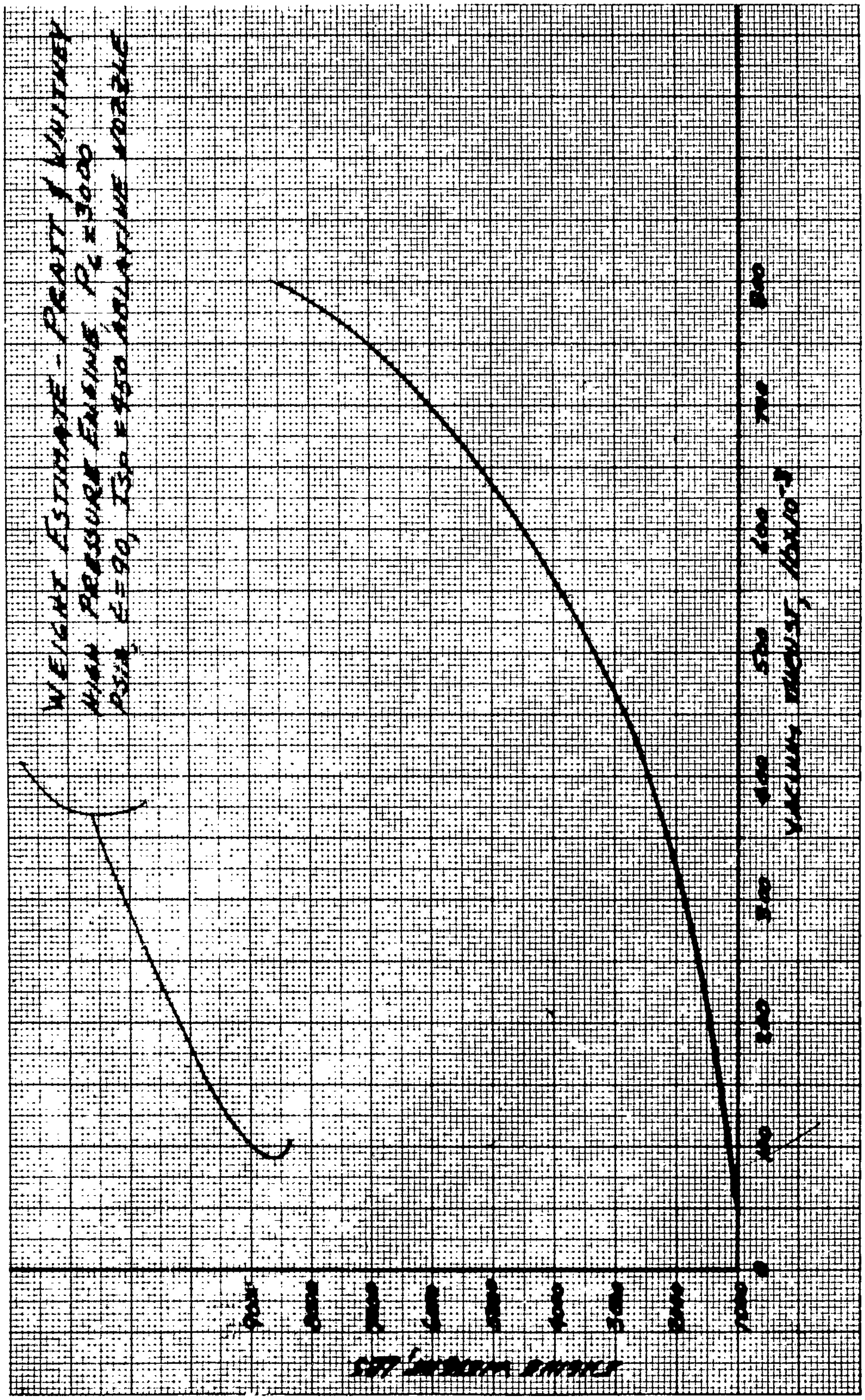


Figure 66. Weight Estimate - Pratt & Whitney High-Pressure Engine

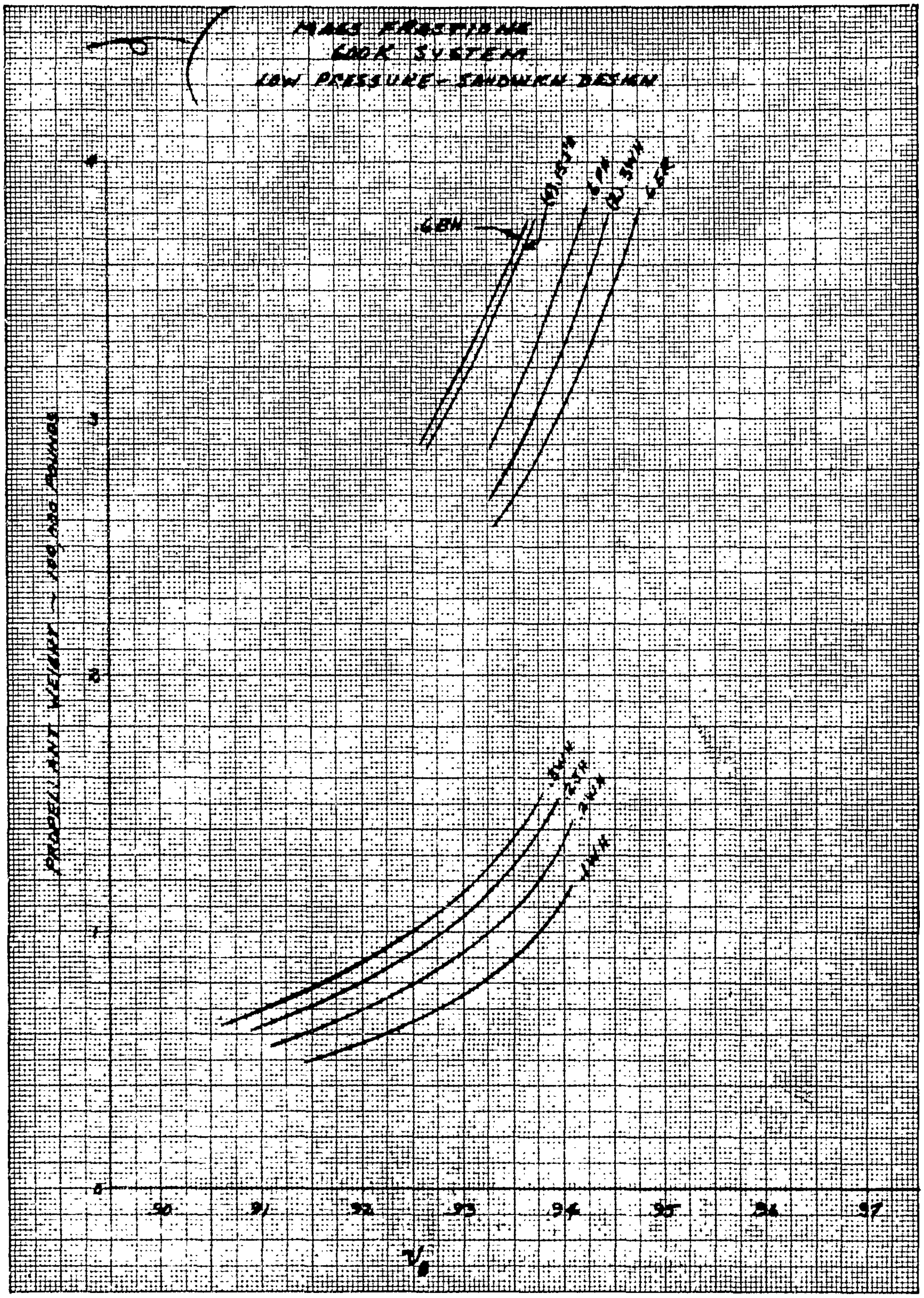


Figure 67. Mass Fractions for 600K System Low-Pressure Sandwich Design

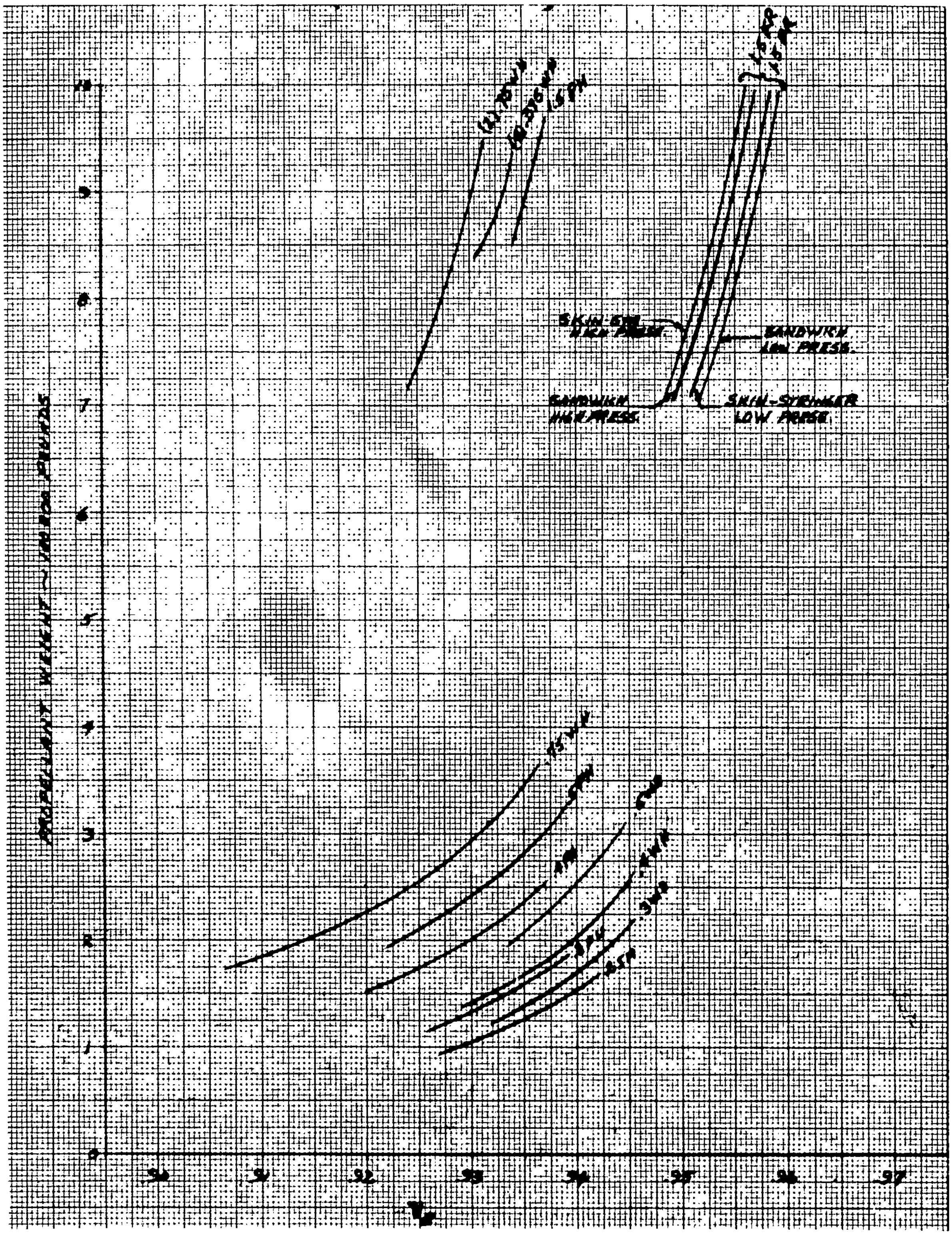


Figure 68. Mass Fractions for 1500K System Low-Pressure Sandwich Design

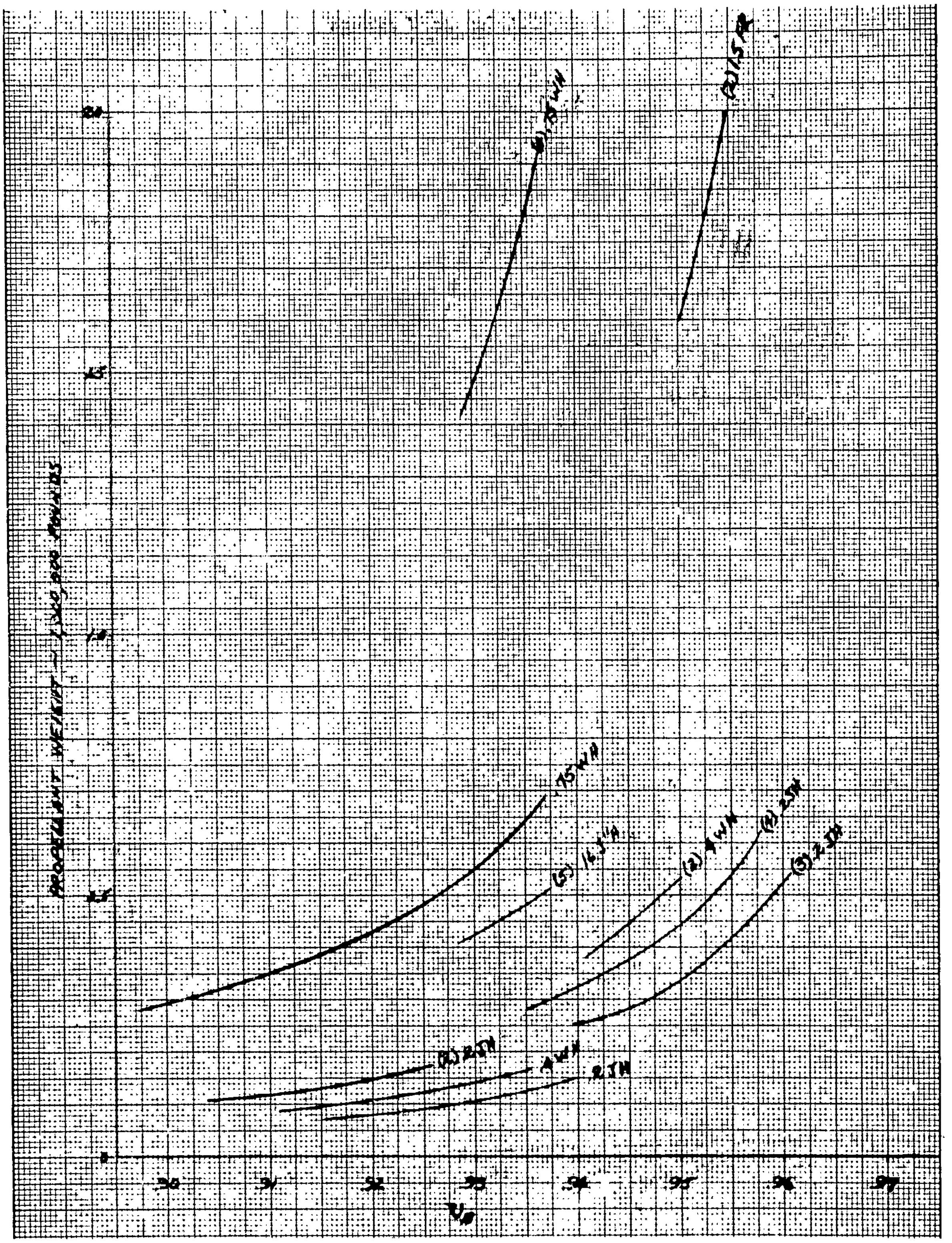


Figure 69. Mass Fractions for 800K and 3000K Systems Low-Pressure Sandwich Design

MACHINE PROGRAM

A typical booster configuration was established in the machine program as a base point. A weight statement and production cost data were obtained for each configuration that had one or more of the following input variables:

- Engine Data
  - Number required
  - Thrust level per engine
  - Weight-each
- Propellant-mass fractions-each stage
- Clustering weight penalty
- Recovery system
  - Wing loading
  - Velocity factor
  - Unit weight of lifting and control surfaces
  - Surface loading power functions
  - Landing gear unit weight
- Propellant unit weight
- Production cost parameters

The weight statements enclosed in the appendix are those of the machine program for the selected configurations. Typical input data statements are inserted for information only. The payload weights are only representative in order to compare the efficiency of the various configurations.

TWO-STAGE BOOSTER SYSTEM

Of the many variables for a given thrust level, the most significant occurred in engine parameters and propellant types. Basically, three types of engines were studied: Bell-nozzle, high-pressure bell (P_c 3000), and plug-nozzle. Propellant types were LO_2-RP-1, LO_2-LH_2, and LF_2-LH_2.

For some configurations the second-stage engine is also used in the first stage in order to minimize cost. Figures 70 through 73 graphically present the representative payload for each configuration studied.

The selected systems used in the study and which have weight statements presented in the appendix are listed in the following table.

Table 13. Two Stage Booster Systems

Thrust Level	State of Art	Advanced Systems
600K	(4) 0.15 J'H/0.2 JH	(2) 0.3 WH/0.3 W'H
800K	(5) 0.16 J''H/0.2 JH	(2) 0.4 WH/0.4 W'H
1500K	1.5 FR/(2)0.2 JH	(2) 0.75 WH/0.75 W'H
3000K	(2) 1.5 FR/(4)0.2 JH	(4) 0.75 WH/(2)0.75 W'H

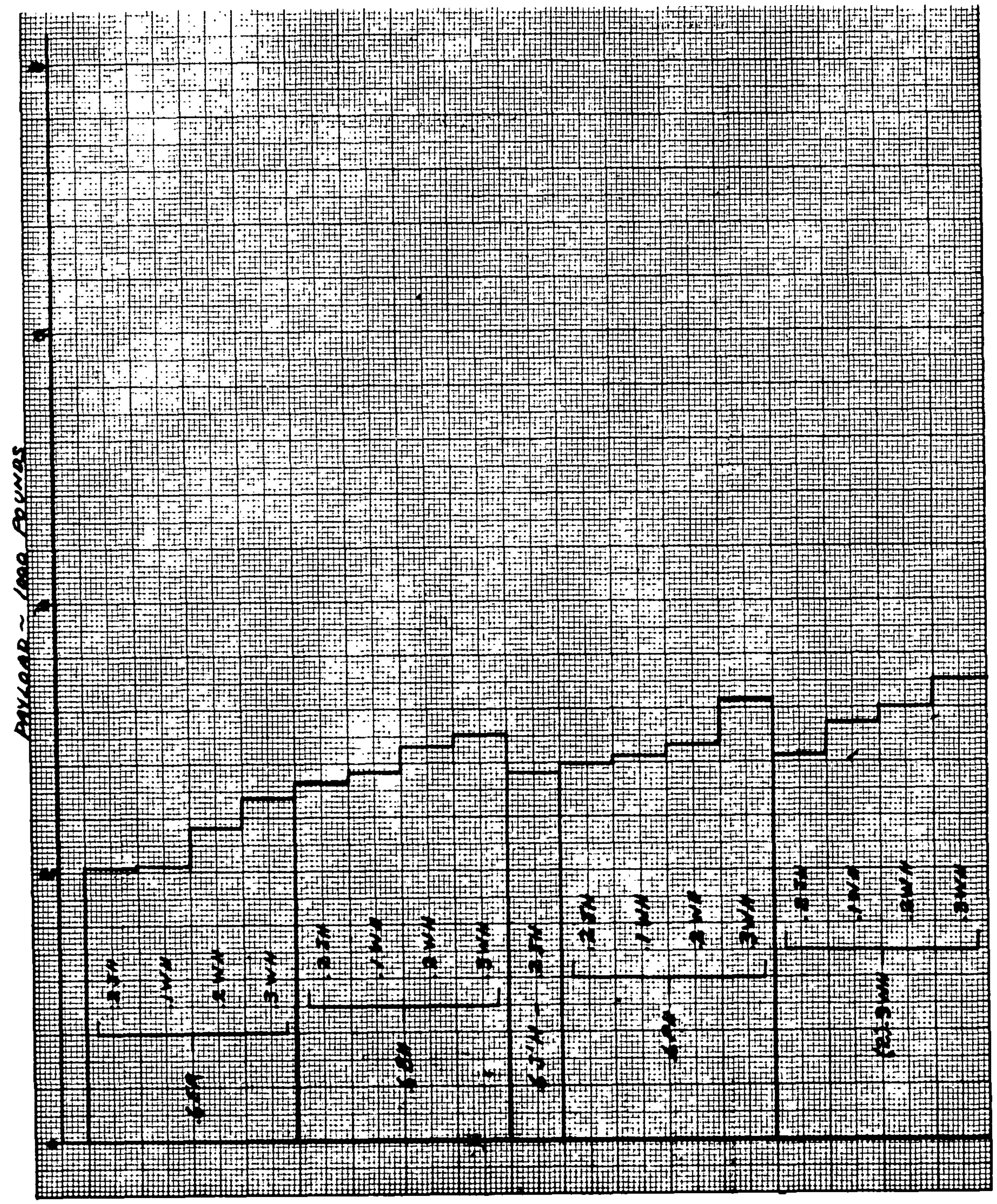


Figure 70. Representative Payloads for 600K Booster Systems

[redacted]

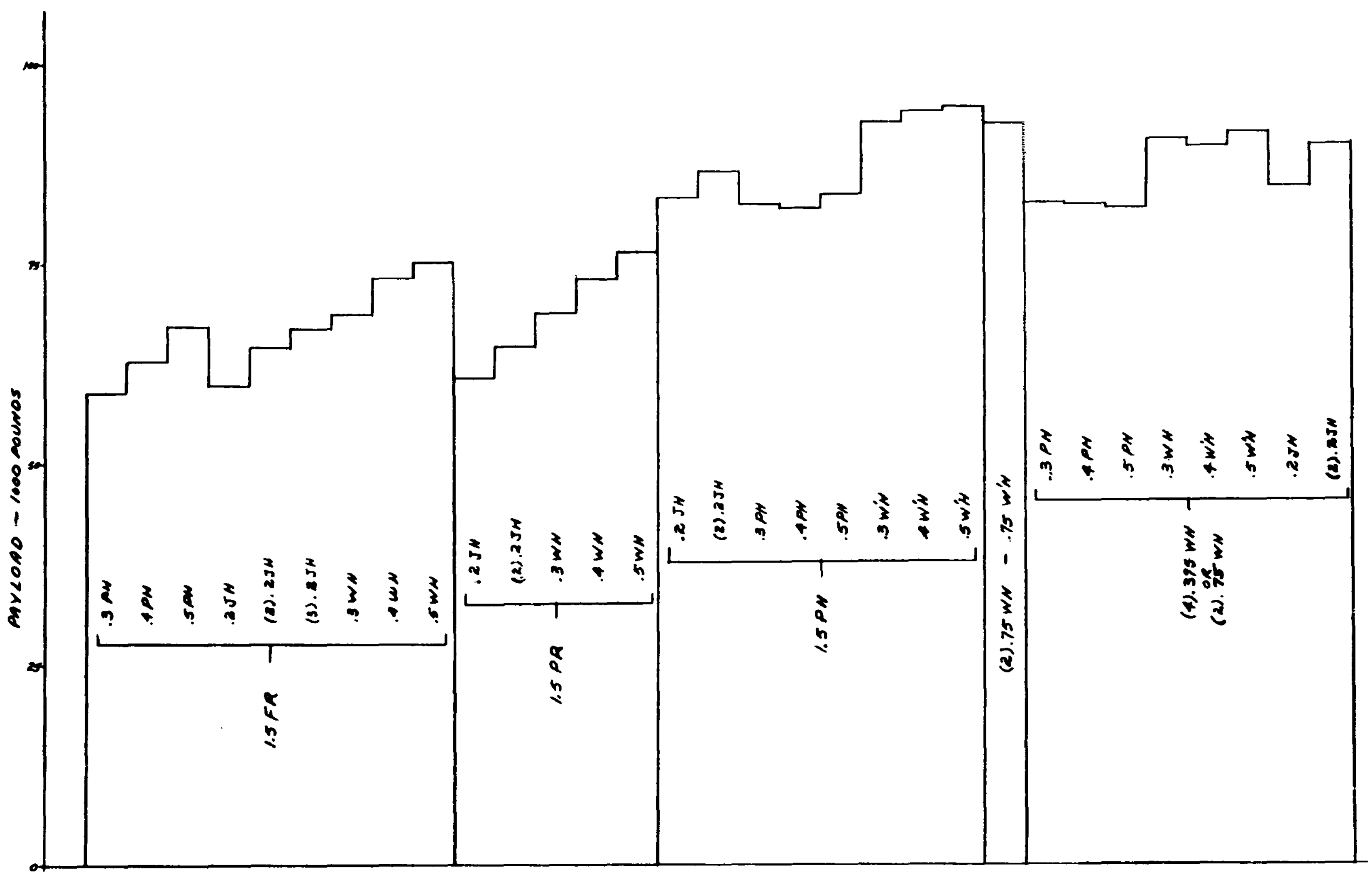


Figure 71. Representative Payloads for 1500K Booster Systems

[redacted]

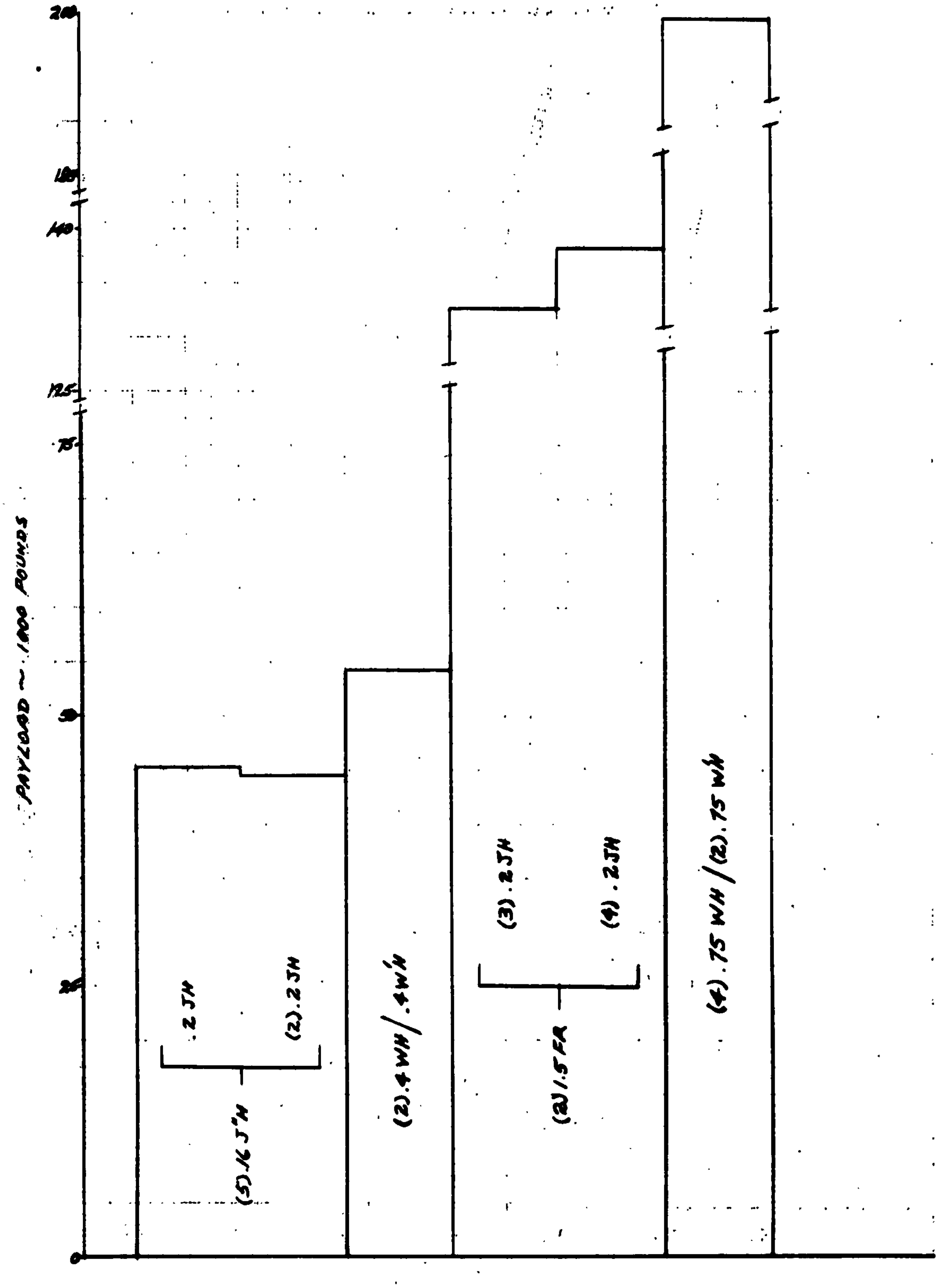


Figure 72. Representative Payloads for 800K and 3000K Booster Systems

[redacted]

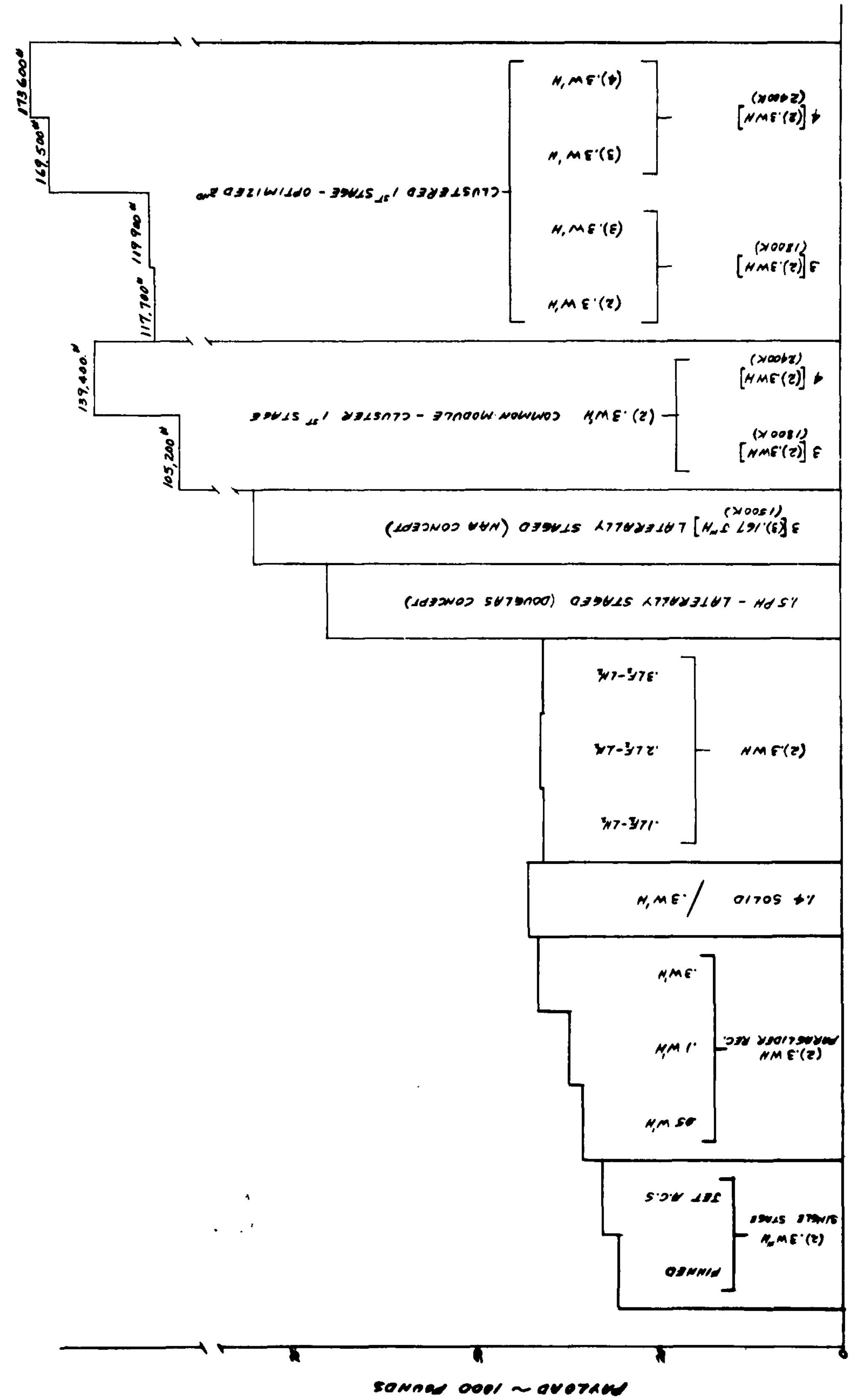


Figure 73. Representative Payloads for Alternate Booster Systems

ALTERNATE BOOSTER SYSTEMS

In order to present a more complete study, several alternate systems were analyzed. Some of these alternate systems were based on the selected systems for each thrust level, others were new systems from which data could be extracted and compared. The representative payloads are graphically illustrated in Figure 73.

The alternate systems analyzed were 600K First-Stage Cluster System, Paragliders First-Stage Recovery, Laterally-Staged Boosters, Segmented Solid First Stage, LF_2/LH_2 - Second Stage, and Single-Stage-To-Orbit.

600K FIRST-STAGE CLUSTER SYSTEM

Based on the advanced selected 600K system, the two clustered first-stage configurations studied were common modules for first and second stages and modular first stage and optimized second stage.

The common-module configuration produced the greatest weight penalty, since it included the second-stage design criteria and the first-stage clustering penalties. The clustering penalties are intermodule attach frames, propellant intertank manifolding provisions, and interstage structure-attachment provisions. Only the clustering provisions are added to the first-stage module for the modular first stage and optimized second-stage configuration. Additional items necessary to cluster both types of modules include longitudinal and lateral ties. Propellant manifolds and an interstage structure that does not lend itself to an efficient design.

Clusters of three and four modules, for the first stage, were studied for both configurations. Thrust levels of 600K, 900K and 1200K were studied for the optimized second stage.

PARAGLIDER RECOVERY OF FIRST STAGE

Paraglider recovery studies were conducted on 600K and 1500K selected designs. A further study was conducted on an extended 600K first stage using a 50K and 100K second stage respectively.

A few of the design characteristics employed for paraglider analysis were a rigid boom, 15 pound per square foot wing loading, landing gear system including a forward wheel and aft skids, and a jet-nozzle attitude control system.

LATERALLY STAGED BOOSTERS

An AMPR weight statement for the Douglas design was prepared for costing. The North American Aviation lateral stage design uses 9 J-2 type engines. Each of the three tanks had a three-engine installation. Two tanks and six engines were dropped at the end of first-stage burning. The analysis resulted in the data shown in Table 14 with those of the Douglas design for comparison.

Table 14. Laterally Staged Boosters Analysis Data

Design	Payload Weight	Gross Weight	Propellant Weight	AMPR Weight
North American Aviation	77,000	1,177,000	1,040,000	37,800
Douglas Aircraft Company	70,000	1,177,000	1,046,245	45,855

SEGMENTED SOLID FIRST STAGE

The solid-rocket weight includes the structure for clustering skirts and thrust vector controls with the rocket case as in the data supplied by the solid-rocket suppliers. Vender supplied data was employed in the design analysis. The interstage weight is based on North American Aviation design and analyses. The resulting weights for this design are summarized as follows:

Item	Weight
Payload	42,700
Gross	728,350
Propellant - Solid Stage I	409,240
Propellant - Liquid Stage II	208,000
AMPR - Stage I	55,810 (Rocket Engines less Propellant)
AMPR - Stage II	9,100

LIQUID FLUORINE/LIQUID HYDROGEN SECOND STAGE

The LF_2/LH_2 - second stage was analyzed, coupled with the selected 600K first-stage advanced systems design. The orbital payload weight for this system approximated that of the advanced selected design, but the associated production cost precluded any further study of this system.

SINGLE-STAGE-TO-ORBIT

The preliminary weight estimate for two designs are summarized in the following table.

Table 15. Attitude Control System

Design	Payload Weight	Gross Weight	Propellant Weight	AMPR Weight
Pure Jet	32,850	500,000	436,000	13,000
Fins Plus Jet	30,740	500,000	436,000	16,000

The structure is generally aluminum-faced sandwich construction. The payload weight quoted is based on an estimated three sigma error analyses assuming a normal distribution. This reduces the payload weight by about 10 percent. This error analyses covers only the weight empty items since propellant density and weight variations can only be considered in an overall weight performance error analyses.

RELIABILITY

METHODOLOGY

This study is first concerned with the development of a predicted reliability growth curve for a two-stage vehicle; it is derived from an analysis of the demonstrated reliability of several current missiles. This vehicle reliability will be used as an example of the values expected for a two-stage vehicle in the period 1965 to 1975, based on the program concepts maintained for the current missiles. An analysis will be conducted to determine whether this predicted reliability results in unreasonably high vehicle replacement costs (because of the unreliability and the number of launches contemplated). Once a reasonable reliability growth pattern has been predicted for the example two-stage vehicle, objectives will be established for the other vehicle configurations under study.

A comparison will be made of each vehicle configuration based on an average reliability for the 10-year operational program.

GROUND RULES

Several ground rules governing the basic assumptions and criteria utilized throughout this study are discussed in the following paragraphs.

Chemical Rocket-Engine Reliability

The single-engine reliability predictions used for all calculations are shown in Figure 74 and Table 16. Rocketdyne has established a reliability of 99.9 percent as their single-engine growth objective. For the purpose of this study, this value has been considered as the 10th operational year reliability objective for all single engines being considered.

Solid Rocket Motor Reliability

Data obtained from Aerojet General Corporation and Thiokol Chemical Corporation were utilized for the reliability predictions of the solid first-stage vehicle configuration.

[redacted]

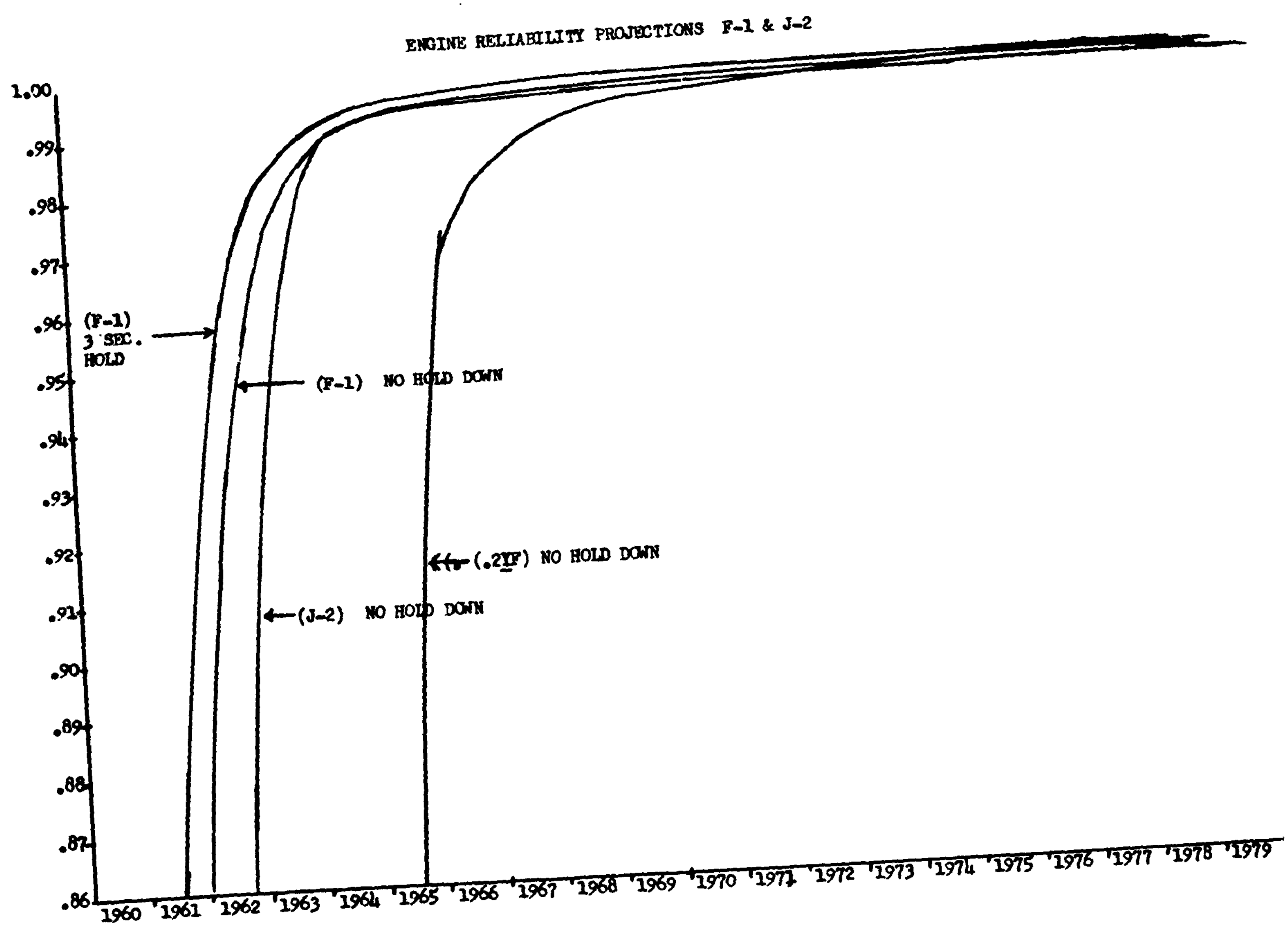


Figure 74. Engine Reliability Projections (F-1 and J-2)

[redacted]

Table 16. Engine Reliability Predictions

	Thrust (lbs)	10-Year Average (%) Engine Reliability	Time To PFRT From Contract Award (Months)	Reliability At PFRT (%)
PMA	25,000	99.00	28	99.00
	50.000	99.00	30	99.00
	75,000	99.00	31-1/2	99.00
	100,000	99.00	32-1/2	99.00
	150,000	99.00	34-1/2	99.00
	200,000	99.00	36	99.00
	300,000	99.00	38	99.00
	400,000	99.00	40	99.00
	500,000	99.00	41-1/2	99.00
F-1	1,500,000	99.72	48	98.00
J-2	200,000	99.68	36	97.50
E-1	600,000	99.72	36	
GE	600,000	97.00	45	97.00
	1,500,000	97.00	45	97.00

System Reliability Predictions

System reliability predictions were based on the distribution of system failure causes from missile and space system flights and from analyses of vehicles similar to the ones evaluated in this study. From this failure distribution, the percentage of failures attributed to each vehicle system was determined. These percentages were used as a measure of their relative operational complexity to apportion total vehicle reliabilities to each internal system.

Confidence Development

Confidence that the attained reliability at the first vehicle launch equals or exceeds the predicted value must be developed through a component and system confidence development program.

[redacted]

DEMONSTRATED RELIABILITY

The development of reliability growth curves for the cost optimized boosters posed three general questions: (1) What initial reliability and type of improvement should be expected for the period 1965 to 1975? (2) If the same reliability growth obtained by some of the current missile programs is assumed for the boosters under study, what will be the resultant initial reliability and rate of growth? (3) If the predicted reliability for the vehicle is determined to be unacceptable because of manned, economic, or national prestige considerations, what objectives would be established?

In order to answer the first two questions, an analysis of the reliability growth patterns of the Thor, Atlas, Jupiter, and Polaris (first stage) missiles was conducted. Flight test data from each program were obtained and plotted by means of a 9-month moving average. Figure 75 indicates the demonstrated reliability of each of these vehicles plotted against calendar time. These curves were averaged and the values plotted to illustrate the reliability of a composite single-stage vehicle during the calendar period. Since Cost Optimized Booster Systems Study is concerned with two-stage vehicles, the single-stage composite vehicle reliability values were squared to provide an estimate for a two-stage vehicle. It is significant to note that the programs initiated at later dates (Thor and Polaris) have higher rates of growth than the earlier (Atlas and Jupiter) programs.

In Figure 76, the same reliability curves are plotted to indicate how reliability varies as a function of the number of years from contract award. These curves demonstrate the reduction in time-to-first-launch for the programs with later calendar start dates. Of particular interest is the reliabilities demonstrated after approximately 36 months from contract award, since this is the anticipated time from contract award to first launch for the vehicles under study. It should be noted (Figure 76) that the reliabilities of the Thor and Polaris programs are higher (36 months from contract award) than the Atlas and Jupiter.

The higher absolute reliability values and rates of growth demonstrated by these more recent programs can be attributed to improvements in state of the art, as well as greater personnel skills. The composite improvement will be defined as improvement in systems technology. The problem now is to extrapolate observed improvements in systems technology to the point in time when the first cost optimized booster is scheduled to be launched.

If the last three and one-half years of the composite curves shown in Figures 75 and 76 are compared, the average rate of growth in systems technology can be determined as follows:

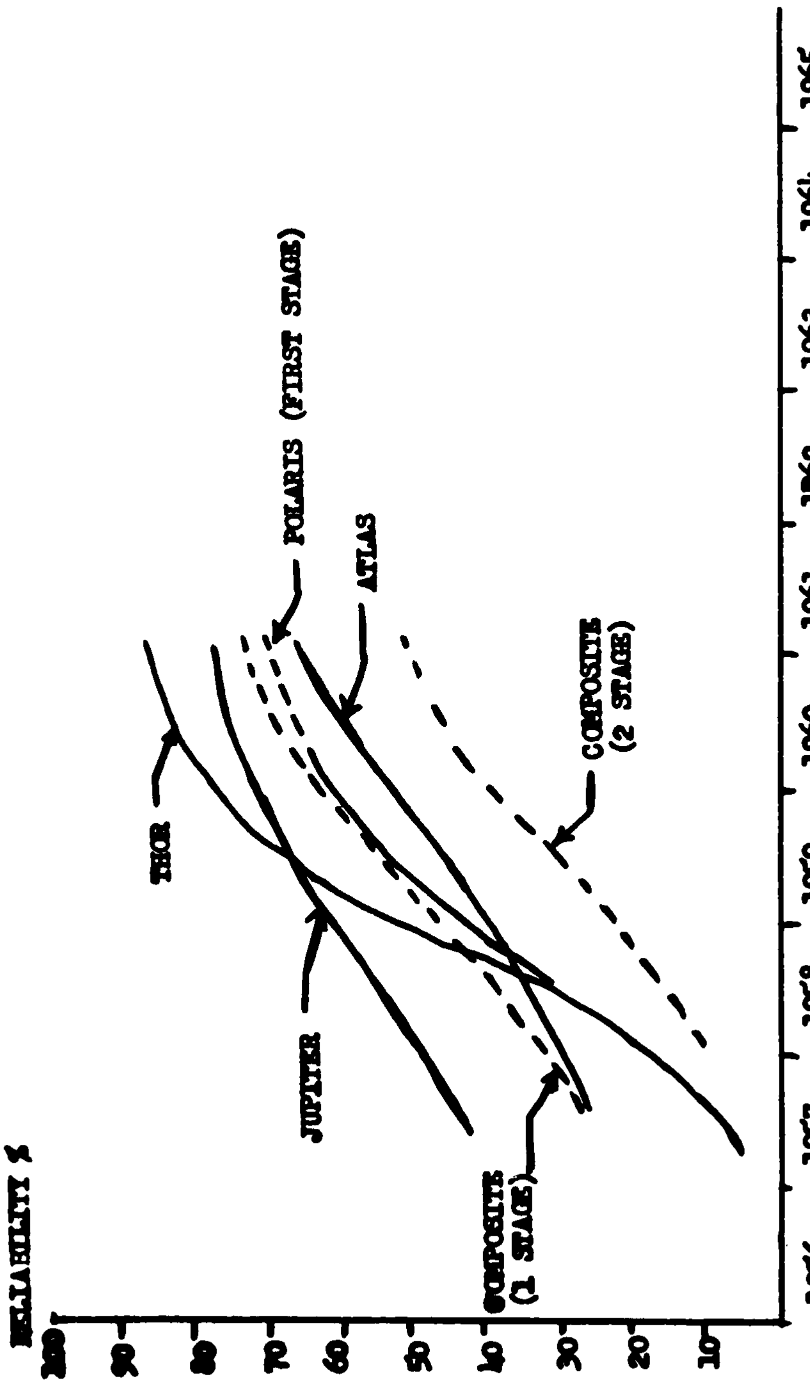


Figure 75. Demonstrated Reliability

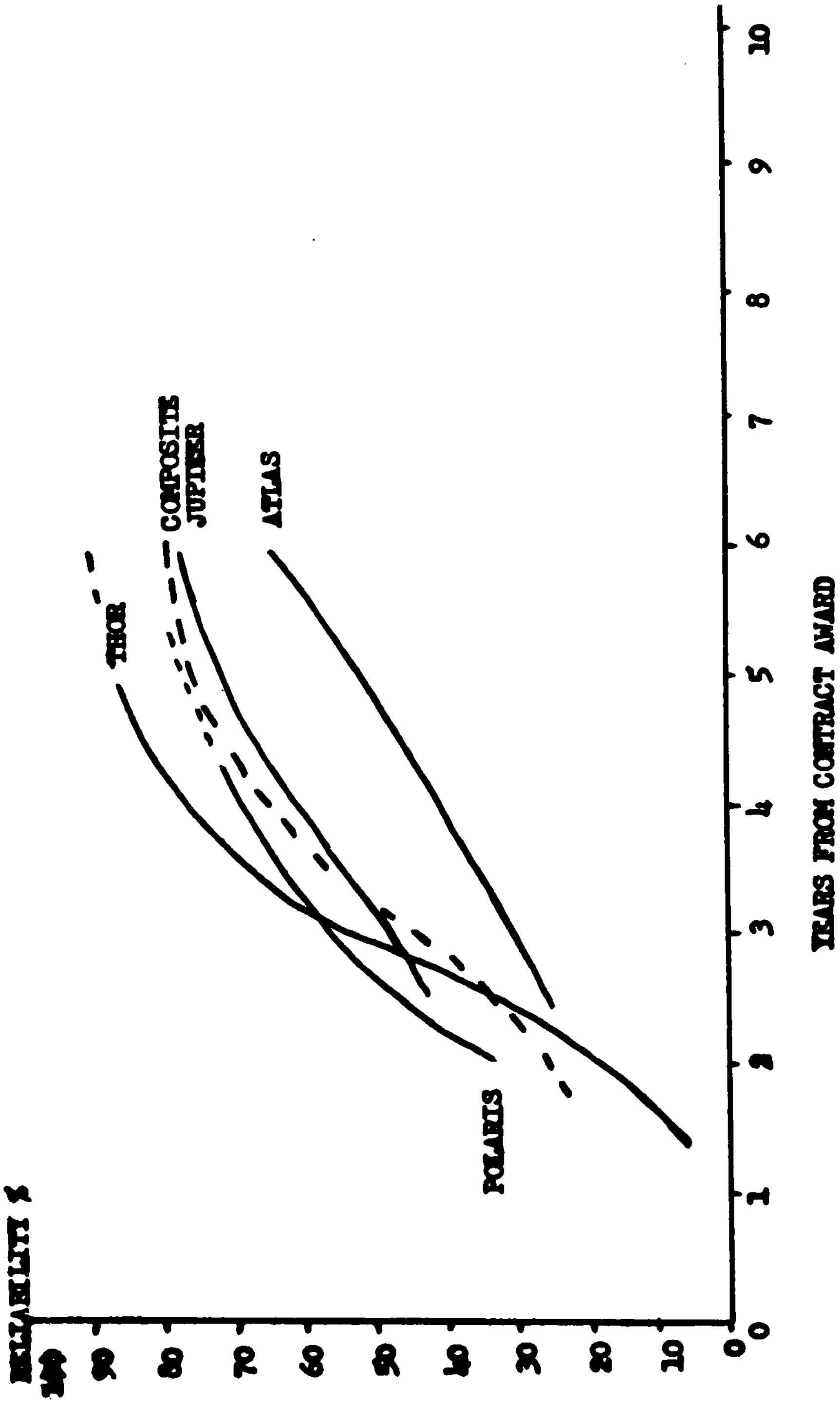


Figure 76. Reliability Growth of Systems

1. From Figure 75: Reliability improvement per calendar year

$$= \frac{0.75 - 0.26}{1960 - 1957.5} = 0.1430$$

2. From Figure 76: Reliability improvement per year from contract award

$$= \frac{0.80 - 0.38}{76.0 - 2.5} = 0.1199$$

3. The rate of improvement in systems technology

$$= 0.1430 - 0.1199, \text{ or } 2.31 \text{ percent per year.}$$

Referring to Figure 76, the composite vehicle is 50 percent reliable 36 months from contract award (March 1959). The rate of improvement in systems technology (2.31 percent per year) was multiplied by the number of years from March 1959 to the anticipated first booster launch at the end of the first quarter of 1966 (seven years). This value, added to the 50 percent, results in a predicted reliability (for a single-stage vehicle in 1966) of approximately 66.17 percent. Applying the 2.31 percent per year to the Thor reliability, which is probably more realistic than the composite vehicle, the predicted reliability at the time of the first cost optimized booster launch is 76.7 percent. A two-stage vehicle (similar to two Thors) then would be approximately $(76.7)^2$, or 58.9 percent reliable.

Now that the predicted reliability for the first launch of a typical two-stage cost optimized booster has been established, a rate of reliability improvement can be applied to this initial value in order to develop a reliability growth pattern. The composite vehicle growth curve shown in Figure 76 demonstrates the expected rate of reliability improvement (approximately 10 percent per year initially) after the first-booster launch. This composite curve was impressed on the newly derived initial reliability to obtain a first estimate of reliability growth for the vehicle under study (see Figure 77).

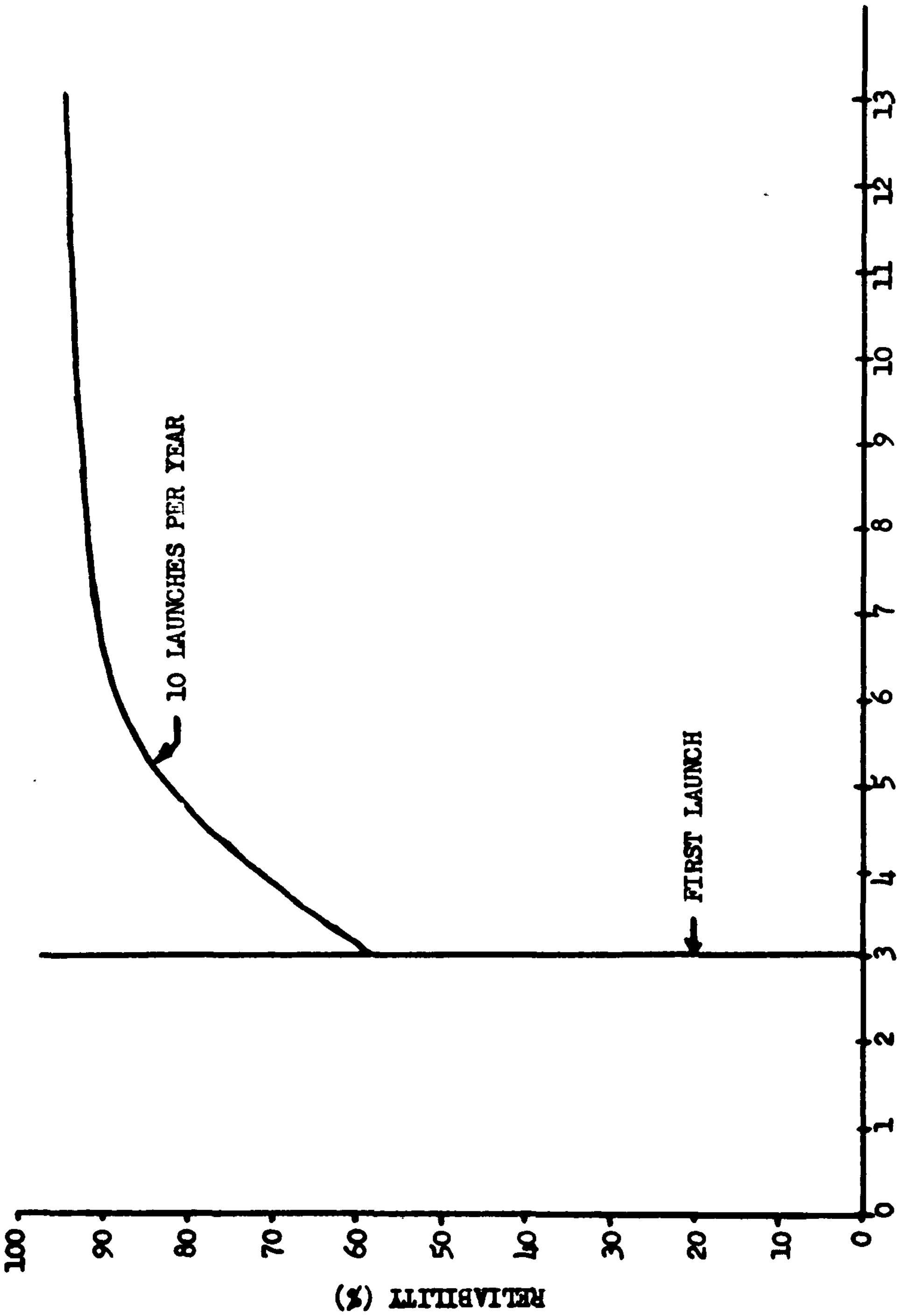


Figure 77. Predicted Reliability Growth for Configuration 1.5FR/2(0.2)JH
(Assumes Normal Funding)

PREDICTED RELIABILITY GROWTH FOR CHEMICAL VEHICLES

The system reliabilities required to meet the overall vehicle reliability objectives established should now be estimated. An analysis of the flight histories of the Atlas, Thor, Jupiter, and Polaris (first stage) provided a distribution of failures for the systems in these vehicles. From this distribution, the percentage of failures attributed to each of the internal systems was determined. These percentages, shown in Table 17, were used as complexity factors (or weighting factors) to apportion to each system the reliability required to meet the overall vehicle objectives. These apportioned values are shown in Table 17 for configuration 1.5FR/2(0.2)JH. This vehicle reliability was selected to represent the general growth pattern of the vehicle under study (based on the program concepts of the current missiles) because of the amount of engine reliability information available and the fact that it was evaluated in both phases of the study. An analysis was conducted to evaluate this reliability, with respect to the number and cost of the replacement vehicles necessary to compensate for the difference in the predicted reliability and a value of 100 percent. The difference between the initial reliability shown in Table 17 and the value of 58.67 predicted for the first launch resulted from a substitution of engine reliability predictions for the original apportioned values.

ENGINE RELIABILITY PREDICTIONS

Engine reliability predictions utilized for all vehicle evaluations are shown in Figure 74 and Table 16. The use of clustered engines poses a serious reliability problem. In clusters, where all engines must operate satisfactorily to achieve successful boost, the engine reliability is the product of the values of each single engine (reliability product rule). Therefore, for a cluster of four engines, each with a reliability of 0.99, the total reliability is 0.99^4 or 0.96.

Engine-Out Concept

Large booster studies conducted at S&ID have indicated that it is desirable to provide the capability to shut down a malfunctioning engine (when clusters of four or more are considered) during predetermined phases of the boost mission. This allows a transfer of the unused oxidizer and fuel to the remaining engines. This concept is known as "engine-out." Utilizing this engine-out concept and the associated reliability equations produces a four-engine cluster reliability of 0.989 (or a single engine reliability of 0.99) compared to the value of 0.96 previously calculated. Although definite advantages are shown with the engine-out capability, the rigorous analysis required to apply the concept to a particular vehicle moves it beyond the scope of this study. For this reason, the cluster reliabilities were computed by the product rule.

Table 17. Reliability Growth Predictions (For Configuration 1.5 FR/2(0.2)JH)
(Assumes Normal Funding)

Item	Complexity Factor	Reliability (Percent) Years From Contract Award					
		3	4	5	7	10	13
Guidance	16.5	91.2	94.6	97.0	98.40	98.72	98.93
Flight control	8.3	95.5	97.2	98.5	99.20	99.36	99.45
Propellant feed	4.1	97.7	98.6	99.2	99.60	99.68	99.73
Propellant utilization	6.6	96.4	97.8	98.8	99.30	99.49	99.57
Electrical	7.4	96.0	97.5	98.6	99.30	99.43	99.51
Pressurization	4.1	97.7	98.6	99.2	99.60	99.68	99.73
Airframe	6.6	96.4	97.3	98.8	99.30	99.49	99.57
Separation	2.5	98.6	99.2	99.5	99.75	99.81	99.84
Rocket engines	2.9	98.8(99.4*)	99.0(99.5*)	99.3(99.7*)	99.6(99.8*)	99.8(99.9*)	99.8(99.9*)
STAGE II RELIABILITY	59.0	72.0	81.8	89.4	94.0	95.4	96.1
Flight control	8.3	95.6	97.2	98.5	99.2	99.36	99.45
Propellant feed	4.1	97.8	98.6	99.2	99.6	99.68	99.73
Propellant utilization	6.6	96.5	97.8	98.8	99.3	99.49	99.57
Pressurization	4.1	97.8	98.6	99.2	99.6	99.68	99.73
Electrical	7.4	96.0	97.5	98.6	99.3	99.43	99.51
Airframe	6.6	96.5	97.3	98.8	99.3	99.49	99.57
Rocket engines	3.7	99.2	99.4	99.5	99.8	99.9	99.9
STAGE I RELIABILITY	40.8	80.9	87.6	92.9	96.4	97.0	97.61
TWO-STAGE VEHICLE RELIABILITY PREDICTION		58.2	71.7	83.1	90.6	92.5	93.8

*Predicted reliability for a single engine

ECONOMIC ANALYSIS OF PREDICTED RELIABILITY GROWTH

The intent of the analysis to this point has been to determine what reliability should be expected for a vehicle based on current missile program concepts. This was accomplished by producing an estimate of the initial reliability and rate of improvement for a cost optimized booster for the period 1965 to 1975. The curve presented in Figure 77 represents this prediction. Although it is considered representative of the type of reliability expected (if the program under study is typical of the Atlas, Thor, Jupiter and Polaris programs) it is not implied that these are the desired reliability levels for the boosters being studied.

In order to answer the third foregoing question (whether the predicted reliability is acceptable, and if not, what objectives should be established) some measure of the merits of the predicted reliability had to be defined which would allow an evaluation of possible alternate objectives. The expected number of mission failures resulting from the difference between the predicted reliability and a value of 100 percent was selected as this measure. For a particular launch rate, vehicle cost, and payload cost, the mission failures can be converted into dollar losses. The approximate number of unsuccessful missions, for the reliability growth curve shown in Figure 77, is the product of the launch rate over a specified period and the average unreliability (1 - R) during that interval. The number of unsuccessful missions for the program with 100 launches was determined and the results are shown in Table 18.

Table 18. Estimated Number Of Unsuccessful Missions For 100 Launches

	Operational Year									
Item	1	2	3	4	5	6	7	8	9	10
Number of launches	10	10	10	10	10	10	10	10	10	10
Average Unreliability	0.35	0.22	0.15	0.11	0.09	0.08	0.075	0.07	0.07	0.065
Approximate number of unsuccessful missions	3.5	2.2	1.5	1.1	0.9	0.8	0.75	0.7	0.7	0.65
10-Year total of unsuccessful missions	13									

In order to provide a basis for an economic comparison of alternate reliability growth curves for this launch program, a fixed payload weight of 64,462 pounds and a cost of $500 per pound was assumed. This payload

cost of $32, 231, 000 was then added to the unit vehicle production cost of $3, 258, 540 (based on 100 vehicles) to produce a total cost of $35, 489, 540. Since the expected losses have been determined to be 13 missions, the vehicle replacement costs will amount to approximately $461, 364, 000 over the 10-year operational program, based on the predicted reliability. This value does not include many other costs that may be directly or indirectly associated with mission failures, which probably make the estimate conservative.

The R & D costs (prior to the first operational launch) for this program amount to approximately $176, 000, 000, exclusive of ground support equipment and engine development costs (the F-1 and J-2 are presently under development). Comparing this cost to the estimated vehicle replacement costs leads to the question of how to reduce the replacement costs without increasing the total program costs. This was answered by constructing a model to illustrate the relationship between the reliability and program costs, as shown in Figure 78.

As indicated in Figure 78, reliability can be improved from point A to point B by redistributing the R & D and vehicle replacement costs. Providing additional reliability effort early in the program (during R & D) allows for strong emphasis during design, where the most efficient use of funds can be made. Determination of the value of reliability at point B, the amount of increased initial funding required to obtain this reliability, and the reduction in number of mission failures resulting, are discussed in the following paragraphs.

Development of a vehicle reliability growth curve that provided a considerable reduction in mission failures, without requiring system reliabilities considered beyond the state of the art for the time of interest, required that several different curves be developed and evaluated. The solid curve presented in Figure 79 represents one of the more attractive curves economically developed to define point B in Figure 78. System reliability objectives required to meet the overall vehicle values, at various points in time, were computed using the apportionment techniques described previously (see Table 19).

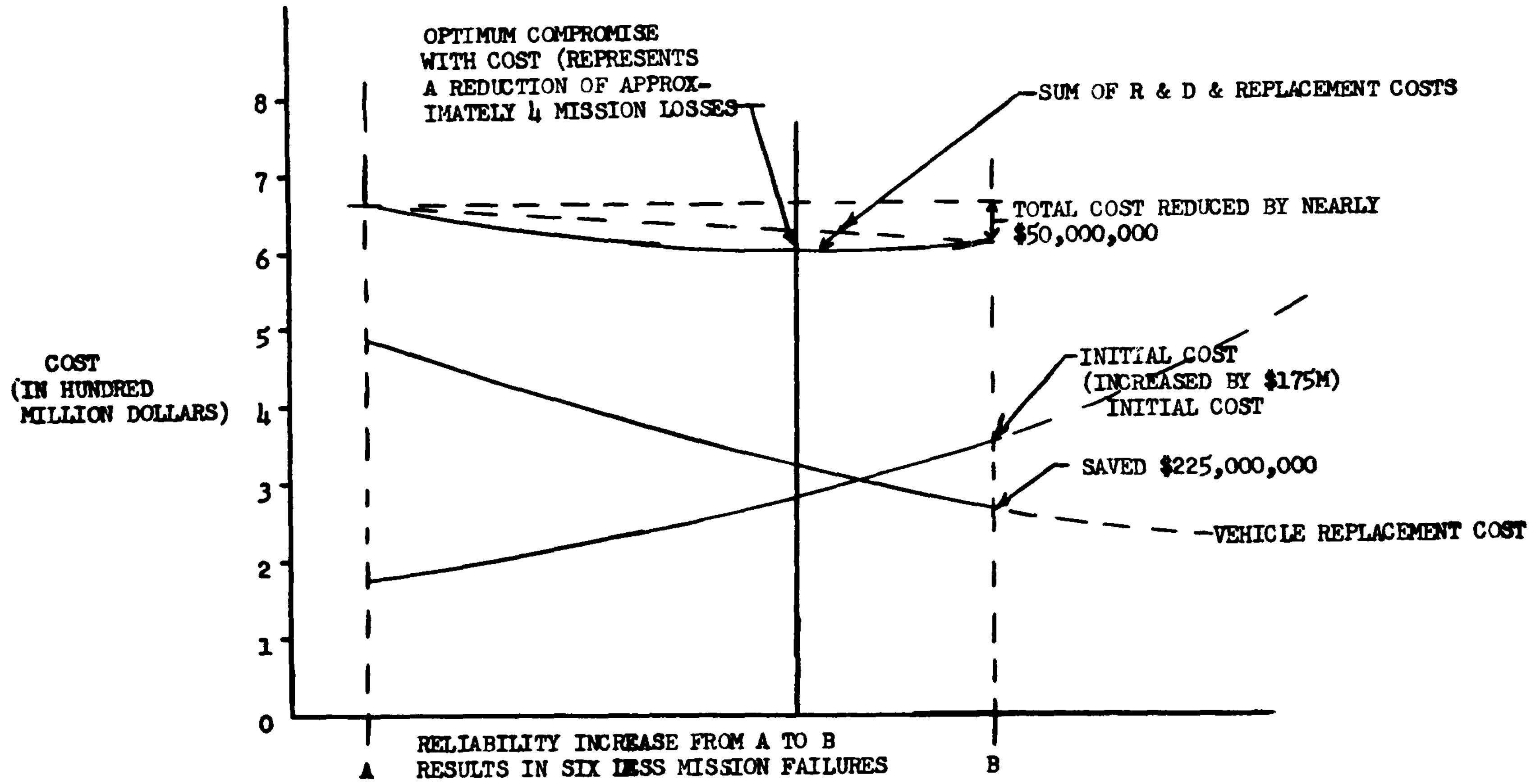
The expected number of unsuccessful missions for this reliability growth curve (Figure 78) was determined to be seven, as shown in Table 20.

Table 19. Cost Optimized Booster Configuration 1.5 FR/2(0.2)JH Reliability Growth Objectives

Item	(Assumes Increased Initial Funding and 10 Launches Per Year) Years From Go-Ahead					
	3	4	5	7	10	13
Guidance	96.2	97.5	98.7	99.3	99.35	99.40
Flight controls	*96.9	98.2	99.0	99.3	99.42	99.43
Propellant feed	*99.8	99.83	99.84	99.85	99.85	99.86
Pressurization	*99.8	99.83	99.84	99.85	99.85	99.86
Propellant utilization	98.6	99.1	99.5	99.72	99.73	99.74
Electrical	98.3	98.9	99.4	99.68	99.69	99.71
Airframe	98.6	99.1	99.5	99.71	99.72	99.74
Separation	99.4	99.6	99.8	99.89	99.90	99.90
Rocket engines	98.8(99.4**)	99.0(99.5**)	99.3(99.7**)	99.6(99.8**)	99.8(99.9**)	99.8(99.9**)
STAGE II RELIABILITY	87.2	91.4	94.4	96.6	97.4	97.6
Flight controls	*96.9	98.2	99.0	99.3	99.42	99.43
Propellant feed	*99.8	99.83	99.84	99.85	99.85	99.86
Pressurization	*99.8	99.83	99.84	99.85	99.85	99.86
Propellant utilization	98.6	99.1	99.5	99.72	99.73	99.74
Electrical	98.3	98.9	99.4	99.68	99.69	99.71
Airframe	98.6	99.1	99.5	99.71	99.72	99.74
Rocket engine	99.2	99.4	99.5	99.8	99.9	99.9
STAGE I RELIABILITY	91.5	94.5	96.5	98.0	98.2	98.2
VEHICLE RELIABILITY	79.8	86.4	91.1	94.7	95.7	95.8

*Predicted reliability based on S&ID studies of similar vehicle systems. Other system reliabilities are apportioned minimum values required to meet the overall vehicle reliability objectives.

**Single engine predicted reliability.



NOTE: THIS FIGURE WAS DEVELOPED FOR A 10 VEHICLE PER YEAR LAUNCH PROGRAM

Figure 78. Reliability-Cost for Configuration 1.5FR/2(0.2)JH

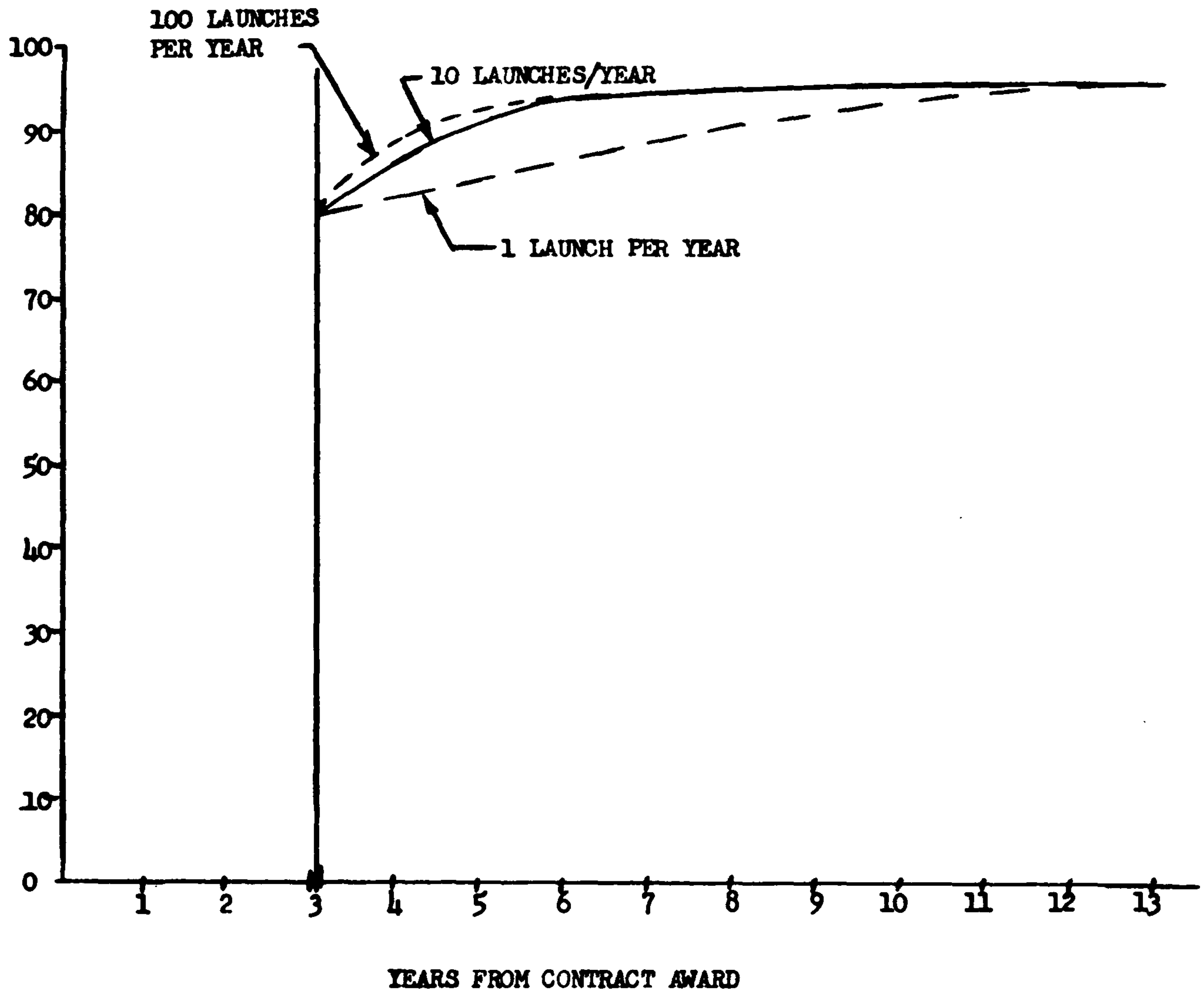


Figure 79. Reliability Growth Objectives for Vehicle Configuration 1.5FR/2(0.2)JH (Assumes Increased R&D Funding)

Table 20. Estimated Number of Unsuccessful Missions for 100 Launches (Based on Predicted Reliability Resulting From Increased R & D Reliability Effort)

	Operational Year									
Item	1	2	3	4	5	6	7	8	9	10
Number of launches	10	10	10	10	10	10	10	10	10	10
Average unreliability	0.17	0.11	0.07	0.06	0.05	0.05	0.045	0.042	0.04	0.04
Approximate number of unsuccessful missions	1.7	1.1	0.7	0.6	0.5	0.5	0.45	0.42	0.4	0.4
10-Year Total of Unsuccessful Missions: 7										

This represents a reduction of six mission failures from the reliability prediction that established point A on the reliability cost model. By converting this savings into dollars, the replacement costs for the reliability at point B was established. The assumption was made that an increase in R & D funds of approximately $175,000,000 would be required to improve the reliability from A to B. However, the savings in vehicle replacement costs amounted to nearly $225,000,000, all of which could be spent during R & D without increasing the total costs shown at point A. Considerable emphasis is often placed on the importance of optimum reliability with respect to cost, without giving adequate consideration to the possible consequences of the reliability at that point. For example, the analysis shows that the optimum reliability-cost lies between points A and B in Figure 78. This indicates that the growth curve established for the increased R & D funding concept is above optimum.

The significant difference between the optimum reliability and the value at point B may not be the cost (which is $25 million higher at B). However, the prevention of two more mission failures at point B (determined by establishing a linear relationship between the reliabilities at points A and B and the distance between them shown on the model) is of major importance and should considerably outweigh the slight cost advantage expected at the optimum value. This reliability-cost analysis has pointed out some of the anticipated effects of program funding concepts on reliability. Further study of this subject is required to predict these effects more accurately, and to determine the influence of higher and lower payload costs and launch rates. More detailed reliability analyses of the selected systems also should be performed to establish more accurate vehicle reliability estimates.

These are some of the areas that could not be thoroughly evaluated at this time, but are recommended for possible follow-on studies. There are methods of improving reliability other than by providing increased funding during an R & D program. The reliability growth shown in Figure 77 is the result of some emphasis placed upon improvement during the operational program. Improvement from the initial reliability to some value on the curve requires considerable effort and expenditures during the operational program. It also takes time, which means that mission failures are occurring because of relatively low reliability. An increase in R & D funds, with proper funding management, can produce the same reliability improvement during design, where the dollars will be more effective.

EFFECT OF LAUNCH RATE ON RELIABILITY

The rate of reliability improvement over the 10-year operational period will be influenced by launch rate. A high-launch rate will disclose more problems in a given period of time than a low rate. To a point, the detection of a greater number of problems allows corrective action to be implemented more rapidly. Theoretically, it may be justifiable to assume that for a launch rate increase of 10 times, reliability can improve commensurately. However, many practical considerations, such as problem identification, reporting, analysis to determine required corrective action, and the inclusion of necessary design changes in the vehicle, should be evaluated. These factors tend to reduce the amount of improvement, and further study is required to predict more accurately the reliability difference that can be realized from accelerated launch schedules.

In order to estimate the influence of the numbers of launches specified in this study (10, 100 and 1000) on reliability growth, it was assumed that (1) 50 percent of the theoretical improvement can be realized for the increase in launch rate from 10 to 100, and (2) 25 percent can be expected for the increase from 100 to 1000. The lower estimate for the latter percentage reflects the influence of launching a vehicle every 3rd day for 10 years, which indicates many vehicles will be launched before design improvement changes can be incorporated. These assumptions were applied to establish the reliability growth curves shown in Figure 79. As indicated, the 1000 vehicle-launch program reliability differs from the 100 launch values only in the initial portion of the growth curve. The rate of reliability improvement shown for the 10-vehicle launch program is anticipated because this extremely low-launch rate will disclose few problems in a given period and, because of the time between launches, many design changes (other than those necessary to correct for problems disclosed during flights) may be incorporated. As a result, each vehicle launched may resemble a new design from a system reliability standpoint.

COMPARATIVE RELIABILITY

During the study, it was necessary to define a single-reliability value to be used in the program cost comparison of each vehicle configuration. The reliability at the median-launch point was utilized for these calculations, since this value closely approximates the average reliability. It also precludes the necessity for calculating average reliabilities for each vehicle configuration. As illustrated in Figure 79, the significant difference in reliability at this point is between the value for 10 launches and 100 launches. This difference, which amounts to 6.3 percent, was used as a constant relationship between these two-launch programs for the comparisons of the remaining vehicle configurations in the study.

VEHICLE RELIABILITY COMPARISONS

To date, this study has concentrated on one vehicle configuration in order to develop relationships between reliability and program funding concepts, and for the effects of the number of vehicle launches on reliability growth. System reliability predictions, based on the concept of increased reliability effort during R & D, have been shown to result in reduced total program costs because of the fewer number of mission failures. Similar system reliability predictions must now be established for each of the other vehicle configurations.

The reliability at the median-launch point (for an assumed equal number of launches per year), as stated, has been determined to approximate closely the average program reliability. Because the study considered 100 vehicle configurations, it was impossible to predict reliability growth curves for each. The procedure used was to determine the engine reliabilities (either clustered or single engines) and the other system reliabilities at the time of the median launch and use this value for the program cost comparisons (see Table 21). In the case of vehicle No. 4, (0.6)WH/(0.2)JH, for example, the first-stage engine reliability at the median-launch point is 99.0 percent and the other systems (similar to those in Table 19) have a reliability of 98.0 percent. The first-stage reliability is the product of these two values, or 97.1 percent. The second-stage reliabilities were computed in a similar manner to provide an estimate of 97.14 percent. The predicted vehicle reliability is the product of these two values, or 94.3 percent. The other vehicle configurations under study were evaluated in a similar manner and the results are presented in Table 21.

Table 21. Vehicle Reliability Comparisons

	Two-Stage Engine Configuration	First Stage Total (%)	Second Stage Total (%)	Product Total (%)
1	(0.6)ER/(0.2)JH	97.76	97.14	94.96
2	(0.6)PH/(0.2)JH	95.09	97.14	92.37
3	(0.6)BH/(0.2)JH	97.76	97.14	94.96
4	(0.6)WH/(0.2)JH	97.05	97.14	94.27
5a	4(0. 5)J'H/(0.2)JH	96.77	97.14	94.00
6a	(0.6)ER/(0.1)WH	97.76	96.48	94.32
6b	(0.6)ER/(0.2)WH	97.76	96.48	94.32
6c	(0.6)ER/(0.3)WH	97.76	96.48	94.32
7a	(0.6)PH/(0.1)WH	95.09	96.48	91.74
7b	(0.6)PH/(0.2)WH	95.09	96.48	91.74
7c	(0.6)PH/(0.3)WH	95.09	96.48	91.74
8a	(0.6)BH/(0.1)WH	97.76	96.48	94.32
8b	(0.6)BH/(0.2)WH	97.76	96.48	94.32
8c	(0.6)BH/(0.3)WH	97.76	96.48	94.32
9a	(0.6)WH/(0.1)WH	97.05	96.48	93.63
9b	(0.6)WH/(0.2)WH	97.05	96.48	93.63
9c	(0.6)WH/(0.3)WH	97.05	96.48	93.63
10a	(1.5)FR/(0.2)JH	97.76	97.14	94.96
10b	(1.5)FR/2-(0.2)JH	97.76	96.83	94.66
10c	(1.5)FR/3-(0.2)JH	97.76	96.52	94.36
10d	(1.5)FR/4-(0.2)JH	97.76	96.22	94.06
11a	(1.5)PR/(0.2)JH	95.09	97.14	92.37
	(1.5)PR/2-(0.2)JH	95.09	96.83	92.08
12a	(1.5)WH/(0.2)JH	97.05	97.14	94.27
12b	(1.5)WH/2-(0.2)JH	97.05	96.83	93.97
13a	(1.5)PH/(0.2)JH	95.09	97.14	92.37
13b	(1.5)PH/2-(0.2)JH	95.09	96.83	92.08
14a	(1.5)FR/(0.3)WH	97.76	96.48	94.32
14b	(1.5)FR/(0.4)WH	97.76	96.48	94.32
14c	(1.5)FR/(0.5)WH	97.76	96.48	94.32

Table 21. Vehicle Reliability Comparisons (Cont)

Two-Stage Engine Configuration	First Stage Total (%)	Second Stage Total (%)	Product Total (%)
15a (1.5)PR/(0.3)WH	95.09	96.48	91.74
15b (1.5)PR/(0.4)WH	95.09	96.48	91.74
15c (1.5)PR/(0.5)WH	95.09	96.48	91.74
16a (1.5)WH/(0.3)WH	97.05	96.48	93.63
16b (1.5)WH/(0.4)WH	97.05	96.48	93.63
16c (1.5)WH/(0.5)WH	97.05	96.48	93.63
17a (1.5)PH/(0.3)WH	95.09	96.48	91.74
17b (1.5)PH/(0.4)WH	95.09	96.48	91.74
17c (1.5)PH/(0.5)WH	95.09	96.48	91.74
18a (1.5)FR/(0.3)PH	97.76	94.53	92.41
18b (1.5)FR/(0.4)PH	97.76	94.53	92.41
18c (1.5)FR/(0.5)PH	97.76	94.53	92.41
6d (0.6)ER/2(0.05)WH	97.76	95.52	93.38
6e (0.6)ER/2(0.10)WH	97.76	95.52	93.38
6f (0.6)ER/2(0.15)WH	97.76	95.52	93.38
6g (0.6)ER/4(0.025)WH	97.76	93.62	91.52
6h (0.6)ER/4(0.05)WH	97.76	93.62	91.52
6i (0.6)ER/4(0.075)WH	97.76	93.62	91.52
7d (0.6)PH/2(0.05)WH	95.09	95.52	90.83
7e (0.6)PH/2(0.10)WH	95.09	95.52	90.83
7f (0.6)PH/2(0.15)WH	95.09	95.52	90.83
7g (0.6)PH/4(0.025)WH	95.09	93.62	89.02
7h (0.6)PH/4(0.05)WH	95.09	93.62	89.02
7i (0.6)PH/4(0.075)WH	95.09	93.62	89.02
8d (0.6)BH/2(0.05)WH	97.76	95.52	93.38
8e (0.6)BH/2(0.10)WH	97.76	95.52	93.38
8f (0.6)BH/2(0.15)WH	97.76	95.52	93.38
8g (0.6)BH/4(0.025)WH	97.76	93.62	91.52
8h (0.6)BH/4(0.05)WH	97.76	93.62	91.52
8i (0.6)BH/4(0.075)WH	97.76	93.62	91.52
9d 2(0.3)WH/2(0.05)WH	96.08	95.52	91.78
9e 2(0.3)WH/2(0.10)WH	96.08	95.52	91.78
9f 2(0.3)WH/2(0.15)WH	96.08	95.52	91.78
9g 2(0.3)WH/4(0.025)WH	96.08	93.62	89.95
9h 2(0.3)WH/4(0.05)WH	96.08	93.62	89.95
9i 2(0.13)WH/4(0.075)WH	96.08	93.62	89.95

Table 21. Vehicle Reliability Comparisons (Cont)

Two-Stage Engine Configuration		First Stage Total (%)	Second Stage Total (%)	Product Total (%)
9j	4(0.15)WH/2(0.05)WH	94.17	95.52	89.95
9k	4(0.15)WH/2(0.10)WH	94.17	95.52	89.95
9l	4(0.15)WH/2(0.15)WH	94.17	95.52	89.95
9m	4(0.15)WH/4(0.025)WH	94.17	93.62	88.16
9n	4(0.15)WH/4(0.05)WH	94.17	93.62	88.16
9o	4(0.15)WH/4(0.075)WH	94.17	93.62	88.16
19a	(1.5)PR/(0.3)PH	95.09	94.53	89.89
19b	(1.5)PR/(0.4)PH	95.09	94.53	89.89
19c	(1.5)PR/(0.5)PH	95.09	94.53	89.89
20a	(1.5)WH/(0.3)PH	97.05	94.53	91.74
20b	(1.5)WH/(0.4)PH	97.05	94.53	91.74
20c	(1.5)WH/(0.5)PH	97.05	94.53	91.74
21a	(1.5)PH/(0.3)PH	95.09	94.53	89.89
21b	(1.5)PH/(0.4)PH	95.09	94.53	89.89
21c	(1.5)PH/(0.5)PH	95.09	94.53	89.89
22	5(0.16)J"H/0.2JH	96.40	97.68	94.16
23	2(0.4)WH/0.4W'H	96.04	97.02	93.18
24	2(1.5)FR/4(0.2)JH	97.45	96.75	94.28
25	4(0.75)WH/2(0.75)W'H	94.12	96.03	90.40
26a	3 2(0.3)WH /2(0.3)W'H	88.91	96.04	85.39
26b	4 2(0.3)WH /2(0.3)W'H	85.40	96.04	82.02
26c	7 2(0.3)WH /2(0.3)W'H	75.68	96.04	72.68
27a	3 2(0.3)WH			
27b	4 2(0.3)WH			
28a	3 2(0.75)WH /2(0.75)W'H	88.91	96.04	85.39
28b	4 2(0.75)WH /2(0.75)W'H	85.40	96.04	82.02
28c	3 2(0.75)WH			
28d	4 2(0.75)WH			
29a	1.5 PH	80.0		
29b	3 3(0.167)J'''H	95.1		
30	Paraglider Recovered 2(0.3)WH/0.3WH	Reference Table 24		
31	2(0.3)WH/0.2XF	96.04	97.4	93.6
32	1.45/0.3W'H (Reference Table 21)			
33	0.6 W'''H(Single Stage)	96.02		96.02
34a	2(0.3)WH*/0.1WH	96.04	97.02	93.19
34b	2(0.3)WH/0.05WH	96.04	97.02	93.19

ALTERNATE BOOSTER SYSTEMS

Laterally Staged Boosters

The laterally staged vehicle concepts developed by North American Aviation and Douglas Aircraft presented many design features for reliability comparison; however, only the more obvious of these can be discussed in this study. A more detailed analysis is required to assess completely all of the advantages and disadvantages of each concept.

The Douglas concept offers a single engine with clustered propellant tanks. The outer tanks separate from the center one after their propellant has been exhausted. Separation of these tanks presents problems because of the size and number of propellant line disconnects. Failure of the main propellant tank disconnects to seal properly after outer tank separation could cause severe leakage. These separation disadvantages may offset the reliability advantages of the single engine.

The North American Aviation concept consists of a cluster of three tanks, each containing three engines. Although more engines are utilized in this concept, each has a reliability of approximately 99.5 percent at the time of first vehicle launch (based on data obtained from Rocketdyne for a 3-second hold capability). The propellant systems of each of these tanks are not interconnected, which eliminates the line disconnects and reduces separation problems considerably. Predicting preliminary reliabilities for these concepts introduced a problem. The reliability value of 80 percent quoted by Douglas was not related to a specific calendar time. It did not state whether the value was an average for some period of time, the initial launch reliability, or an objective for the end of some operational program. The predictions made in the Cost Optimized Booster Systems Study have been an average value for a 10-year program, based on a contract award date of 1963. For this reason a direct comparison of the predicted reliabilities could not be made.

Segmented Solid First Stage

Reliability predictions obtained from several solid motor manufacturers were reviewed for this study. The values quoted differed considerably. In order to present an unbiased estimate of a vehicle with a solid first stage, values of 98 percent received from Aerojet General and 99.6 percent from Thiokol Chemical were used. These two values have been used to provide a North American Aviation estimate and an optimistic estimate of the reliability of this vehicle (See Table 22).

Table 22. Reliability Prediction Vehicle 1.4S/.3W'H

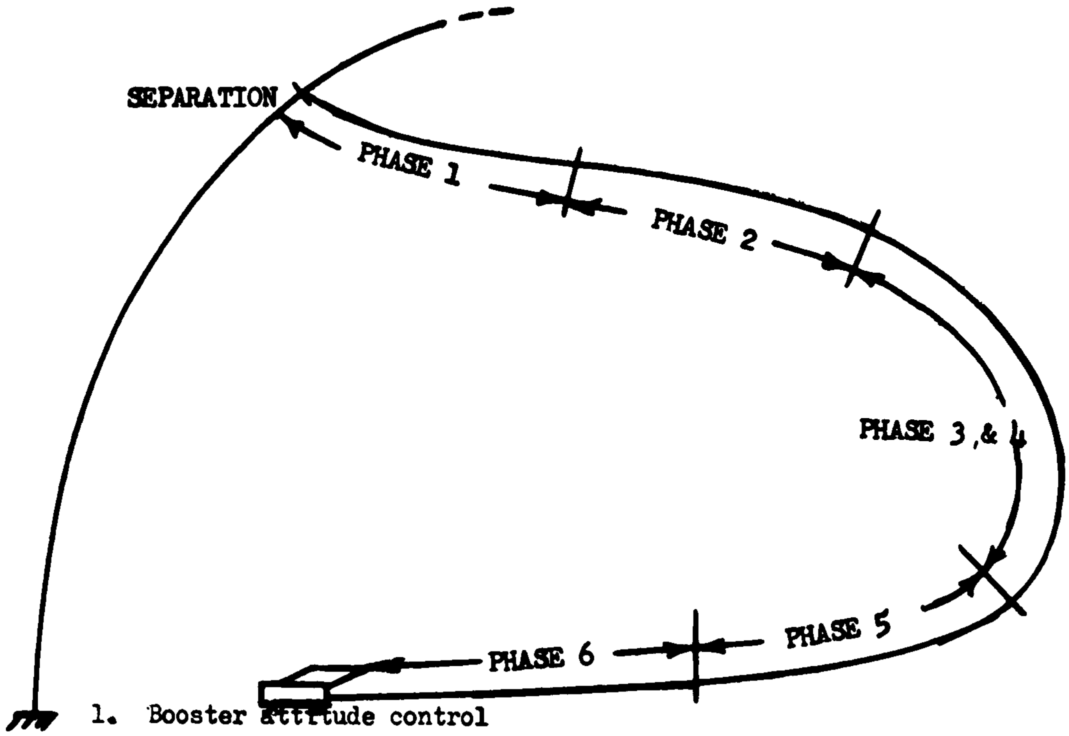
Stage	Reliability (%)
II	97.1
I	
Thrust vector control	99.8
Electrical	99.68
Airframe	99.71
Solid rocket motor	98.0 and 99.6
Stage I Reliability	*97.18 **98.8
Total Vehicle Reliability	*94.4 **95.9
*Predictions using Aerojet data **Predictions using Thiokol data	

Paraglider Recovery

Study of the alternate booster systems included consideration of a paraglider recovery plan for use with the first-stage booster. One of the concepts involved the use of an extended first-stage mission. The other was a conventional mission. The assumption was made that the paraglider would be deployed under approximately the same aerodynamic conditions (through the use of a drogue chute) even though the extended first-stage mission involves supersonic reentry. The probability of successful recovery of the first-stage booster is defined as the product of the boost reliability and the recovery probability of success.

RECOVERY MISSION ANALYSIS

The recovery mission (from the point of booster separation) was divided into six phases as shown in Figure 80. These phases represent major operational events that must occur satisfactorily during a successful recovery mission. The analysis assumes that the booster cannot glide to its landing destination and must be towed by an airplane. The systems required to function during each phase of this mission are defined in Table 23. When recovery system configurations are completed, the required operating time in each mission phase will be utilized in more detailed predictions. Table 24 presents the preliminary system probability of success predictions for each system. These values are minimum requirements to meet the overall mission objectives. Figure 81 is a graphical presentation of the probability of success objectives for the 10-year booster operational program. The probability of success for the first-stage booster recovery is the product of the average boost reliability and the average recovery P_s. For the programs with 100 and 1000 launch missions, the recovery P_s = 96.04 percent x 89 percent = 85.5 percent. For a 10-mission launch program, the recovery P_s = (96.04 - 6.31) x 86.9 percent = 77.98 percent.



1. Booster attitude control
2. Paraglider deployment
3. Glide and guidance to a window in space
4. Engagement and tow by an airplane
5. Final approach and release from tow airplane
6. Flare and landing

Figure 80. Recovery Mission Definition For Reliability Analysis

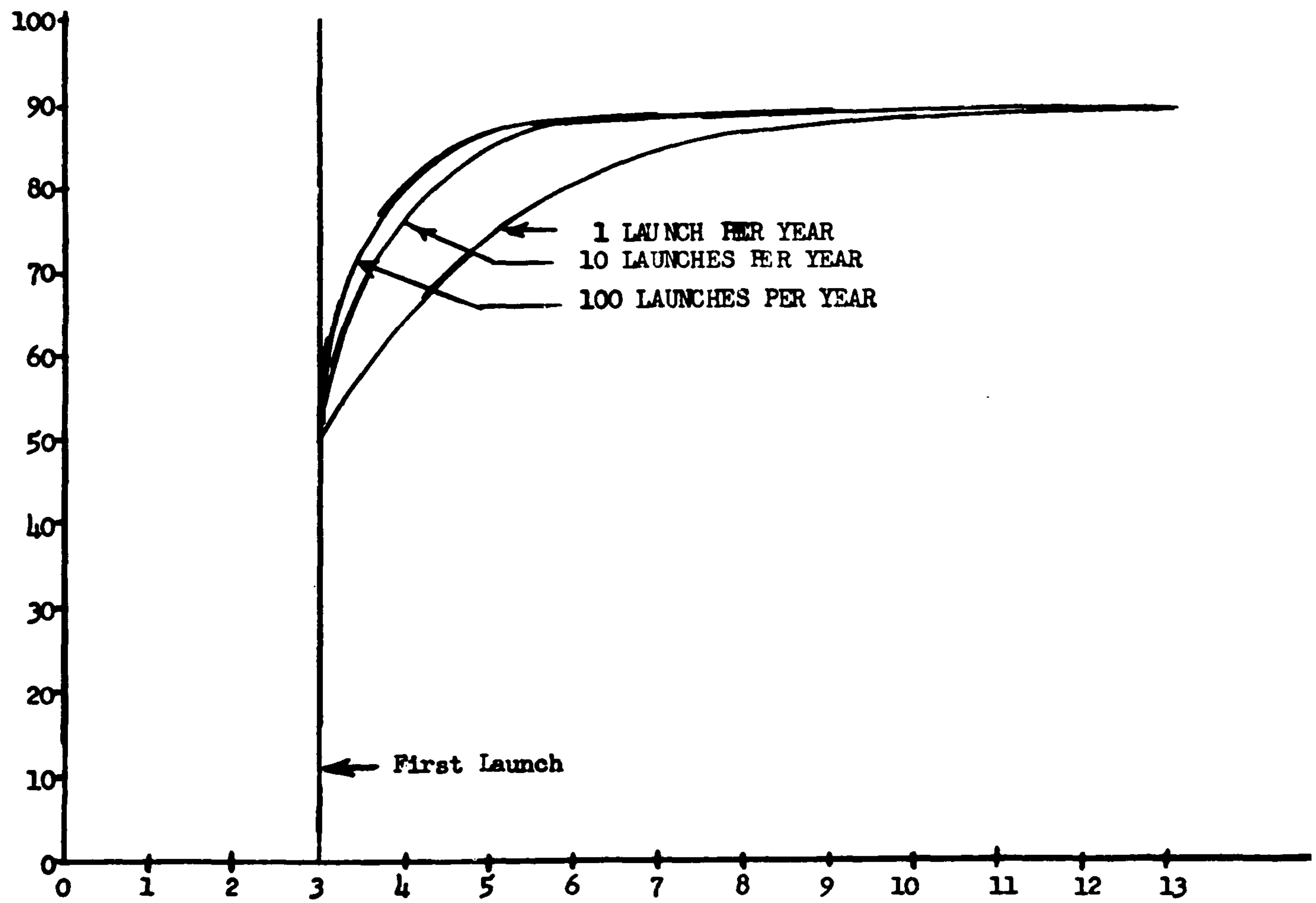


Figure 81. Probability of Successful First-Stage Booster Recovery (After Separation)

Table 23. Booster Recovery Mission

Phase	Systems Functioning
Boost	System warm-up, APU, guidance, and flight control
1. Booster attitude control	Roll stabilization rockets, drogue chute released, *paraglider faring covers removed, flight control
2. Paraglider deployment	*Paraglider release
3 and 4. Maximum L/D glide, powered flight, or *tow airplane engagement	APU, guidance, flight control, turbo-jet
IV. Final approach	APU, guidance, flight control, turbo-jet
V. Flare out	Same as above
VI. Landing	APU, turbo-jet, flight control, landing gear, paraglider release
*Functions occur only with paraglider configurations	

Table 24. Recovery Probability of Success

Systems	Predicted Reliability			
	Years From First Launch			
	0	1	4	10
Guidance	91.5	94.6	98.4	98.9
Tracking beacon				
Radio command receiver-decoder				
Telemeter				
Airspeed sensor				
Flare computer				
Antenna				
Flight Control System	95.5	97.2	99.2	99.5
Auxiliary Power	91.6	93	96	98
Hydraulic system				
Electrical system				
Control Actuation	82.5	95.3	92.1	93.9
Cables				
Landing gear				
Attitude Stabilization	90	93	99	99.1
Pitch				
Yaw				
Roll				
Paraglider Recovery System	59.4	67.8	85.5	89
*Probability of successful engagement by tow airplane	84.5	96	98.9	98.9
Total paraglider recovery probability of success $[P_s(\text{systems}) \times P_s(\text{engagement})]$	50	65	84.5	88

*Probability of successful engagement by a tow aircraft was completed by assuming a single attempt P_s of 50 percent and 4 to 6 attempts. P_s for the total number of attempts was computed by means of the formula: $P_s(\text{total}) = 1 - (1 - P_s)$ for a single attempt.

MANAGEMENT PLAN

The high reliability required for the cost optimized booster is dependent upon strong and positive management direction. Increased R & D funds, as proposed in this report, will not assure reliability attainment without the control and implementation that can be provided by firm management. Effective management requires that reliability activities in all phases of product development be coordinated and directionally compatible. This intraplant direction is achieved at S&ID through a divisional Reliability Policy Board composed of top level division management and headed by the Division President.

The cost optimized booster Project Manager and Reliability Manager will be members of this board, and will be responsible for the direction and control of all aspects of the reliability program and implementation of all policies. The Reliability Manager, reporting to the Project Manager, coordinates with the customer on reliability matters and establishes technical and administrative procedures and methods to meet the program requirements as directed by management and the customer.

The management plan for an effective reliability program, emphasizing design, manufacture, and confidence development testing, is further described in the following paragraphs.

DESIGN

Product reliability begins with design reliability. The reliability criteria is a part of the design criteria at S&ID and it is considered by the design engineer as an important factor in his analysis. Reliability specialists parallel the designers' work offering technical, analytical, and research assistance.

DESIGN CRITERIA

Large design margins are required relative to all functional and environmental stresses, including time as a stress. The design must provide ease in manufacturing the inspection so that workmanship variations will be minimized. The design concept recognizes that both strength and functional characteristics of parts vary due to material and workmanship variations, even under excellent quality control conditions. Design margins and other design features are provided to accommodate such variations to prevent impairment of the functional dependability of the fabricated equipment. S&ID and supplier designs are carefully reviewed for these conditions. Fail-safe features, redundance, and design provisions to minimize the effect of equipment failures are incorporated to the maximum possible extent.

SPECIFICATIONS

SUBSYSTEMS AND COMPONENTS

Each vehicle subsystem and component will be delineated and controlled by an individual booster specification, including existing supplier designs and components covered by existing specifications. Functional and environmental parameters, test criteria, tolerance, reliability requirements and goals, and physical characteristics will be specified for each item.

Each equipment specification and its associated material and process specification will be reviewed and approved by the S&ID reliability group prior to submittal for USAF approval. The reliability review will encompass all factors contributing to reliability strength or resulting in reliability weakness. S&ID procedures include the following characteristics in this review:

1. Functional strength and design margin.
2. Tolerances and system interface tolerability
3. Functional operability and associated human factors (both fabrication and operation).
4. Reliability analysis of failures modes.
5. Reliability requirements (adequacy and applicability).
6. Confidence test requirements (adequacy and applicability as compared to General Test Plan).
7. Materials and Processes.
8. Environmental resistance and effects (natural and induced).
9. Deterioration due to operation, storage or environment.

DESIGN REVIEW

Each functional and structural unit of the vehicle will be processed through a comprehensive reliability analysis and a formal design review

prior to release for manufacture. Nominally, the following three formal design reviews will be required:

1. Preliminary design review, conducted during the layout phase.

2. Major design review, conducted prior to or coincident with the release of R & D drawings.

3. Final design review, at or before 100-percent design release point.

A reliability analysis will be conducted on each component design, including both new or existing supplier designs, each subsystem, and on the combined systems. Designers will participate in this analysis, which will include all materials, detail parts, part application, stresses, margins, tolerances, and failure potentials. Emphasis will be placed on compatibility effects at interfaces at all levels, especially in regard to the tolerability of downstream elements to variations in the output of upstream elements, and to the effect of feedback forces.

A close examination of parts usage and derating factors in relationship to known parts data, critical environmental stresses, and transient conditions also is required. Reliability approval of components selected will be based on experimental evaluation tests, supplier and other contractor tests, prior usage experience, and the results of reliability analysis of the component design and its application in the system. Component selection also will depend upon approval of the source as a 3-million-pound booster supplier. Completion of data and calculations in the reliability analysis will provide an estimate of equipment failure rates, comparison of expected with required failure rates, indication of the principal modes of failure anticipated, data for clear and concise statements as to the acceptance and rejection criteria necessary to ensure that produced items meet performance and reliability requirements, and data for quality assurance planning.

The Design Review Board includes the following regular members (consultant or specialist personnel and customer representatives also attend when necessary):

Reliability Manager, Chairman
Responsible design engineer
Human Factors engineering supervisor
Quality Control representative
Manufacturing representative
Purchasing representative
Contract and Proposals representative

The function of the Design Review Board is to review all aspects of the design, including effect on costs and schedules, and all problems affecting quality, manufacturing operation, maintenance, man and machine integration, or reliability. The review board may approve the design as submitted, may require problem areas to be resolved before approval is granted, or may in certain cases refuse approval and refer the problem to the Program Engineering Manager. When action to resolve a discrepancy or problem is required, the Reliability Manager establishes the responsibility for correction and provides the necessary follow-up. The design is reviewed for approval after the required changes have been made. All Design Review Board actions are documented in S&ID reports which contain copies of the data submitted to the board.

PLANNING AND PROGRAMMING

The reliability program milestones in design, procurement, fabrication testing, and quality areas will be planned and scheduled by the Critical Path Schedule technique utilized at S&ID. Detail milestones from the parts fabrication and processes level are integrated with component, subsystem and vehicle milestones. Testing milestones are programmed in parallel and cross-networks are planned to permit practical integration of test results and assimilation of data generated within a reasonable time span.

REMEDIAL ENGINEERING

Every reliability-significant malfunction that occurs during test, fabrication or checkout will be evaluated by reliability specialists to determine the cause and to assure that corrective and preventive action is taken. The program for effective remedial action requires that positive action be taken to prevent the recurrence of every failure. No failures will be excluded, regardless of the apparent importance or lack of importance. If the probability of recurrence is sufficiently high to affect the reliability of the vehicle, immediate preventive action is required, effective on all existing and unproduced parts. If the probability of recurrence is such that the reliability of the vehicle is not compromised, the preventive action may be made effective at a convenient change point.

QUALITY ASSURANCE

The extremely high reliability associated with the cost optimized boost vehicles demands the highest obtainable quality in all phases of manufacture and assembly. The quality assurance program proposed emphasizes "built-in quality" to an extent far beyond present methods and is designed to reduce the time required to reach high-quality assurance levels.

It is the policy of North American Aviation that there can be no compromise in the quality or safety of its products. All aspects of design, procurement, fabrication, assembly, test, packaging, and delivery are subject to this fundamental concept. It is recognized by S&ID that this concept, as applied to the program, will be difficult to accomplish. However, the approach to be taken, which integrates comprehensive quality, reliability and test plans, provides assurance that these goals will be met and will be in conformance with MIL-Q-9858 and MSFC Quality Engineering Bulletin QEB-NR-2. Quality Control activities are delineated in approximately 145 Division Quality Control Procedures. Each will be reviewed for applicability to the program and submitted for USAF approval at the inception of the program.

PROGRAM FOR MECHANICAL EXCELLENCE

The cost optimized boost vehicle emphasizes control of workmanship and material variables (Quality Control) as well as sorting or culling of defectives (Inspection). To this end, both S&ID and suppliers will be expected to employ craftsmen, technicians, quality engineers, and inspectors of the highest possible caliber.

The Program for Mechanical Excellence is intended to establish and identify the people who produce the hardware, and who are specifically trained to handle space vehicle assignments. The program applies to manufacturing, quality, test, or handling personnel who have personal contact with hardware and thus are responsible for the precision and quality of the individual part and subsystem.

A hard core team of skilled mechanics will be selected by S&ID management for the manufacturing team. Each member will be chartered for a particular assignment and will be given a personal numbered stamp. If he transfers to another assignment, he will surrender his stamp for the assignment he is leaving and qualify for the new assignment. The Program for Mechanical Excellence is reserved for non-management personnel who have physical contact with hardware. An individual who fails to maintain the high standards required by the program is subject to transfer to another assignment.

CONFIDENCE DEVELOPMENT PLAN

An adequate reliability test program, well planned and effectively implemented, is essential to the achievement of the reliability goals. The purpose of such a program is not to test reliability into the equipment but to conduct sufficient tests to verify the results of design effort. Areas of weakness are uncovered early in the program so that controls can be established that will maintain the hardware at the required reliability level. The S&ID program is based on proven methods of designed experimental tests, applied to the lowest part level and carried through to the vehicle system level. Operational life testing, combined environment testing, and system interaction effects are stressed in the Confidence Development Program.

The application approval phase of testing utilizes designed experiments which include combined environment interaction tests, sequentially programmed life tests at significant environmental and functional stress levels, and system interface and interaction tests. The programmed intent of this test phase is the disclosure and elimination of those parts, components, and design quality characteristics which contribute significantly to the risk of system failure. Data is generated to assess the attained reliability and determine the confidence limits. These tests, when satisfactorily completed, provide a basis for approval of parts, components, subsystems, and systems in their application.

The assurance test phase of the Confidence Test Program is the basis for maintaining reliability in the hardware to be utilized for flight vehicles. The criteria for assurance testing is established by analysis of the tests and test results of the application approval test phase. Limitations are placed on environmental exposure during this test phase to preclude wear-out, and destructive or degenerative test effects on flight hardware. The assurance testing is designed to provide data so that a positive measure may be made of flight hardware capability in reference to established goals.

Experience at S&ID on the WS-131B program and at the Rocketdyne Division on J-2 and F-1 rocket engine components has confirmed the validity and effectiveness of the design experiment approach to testing. The S&ID test concept is formulated to integrate all testing completely and attain high confidence in the reliability of the vehicles prior to launch.

GOALS

The Confidence Development Plan proposed for the vehicle integrates all specific test activities, thus eliminating duplication and emphasizing a total inter-related concept to attain the following goals: (1) Component subsystem and system selection and application approval at a high confidence level, (2) maximum test effectiveness at minimum cost through use of designed experiments and comprehensive engineering analysis of all conditions, (3) total utilization and evaluation of all data, and (4) maximum assurance of launch success in regard to man and mission rating.

TEST PLAN

The Confidence Development Plan covers all testing proposed for the development program. Figure 82 illustrates this plan in temporal and test groupings. Testing is grouped into two equipment units—R & D and Flight—with goals of application approval and assurance respectively. Each group is also arranged temporally, indicating the buildup of testing required for each goal to be met. The General Test Plan, a required system data document outlines detail test procedures, sample sizes, equipment requirements, and facilities. The following paragraphs summarize each test grouping and outline basic test requirements and data utilization.

EXPERIMENTAL EVALUATION TESTS

Experimental evaluation tests are performed prior to application approval testing to isolate questionable design criteria, parts, components, and subsystems, and to establish a basis for completing the design. They are primarily engineering functional performance tests using limited environments and mockup or breadboard hardware configurations. Preliminary data are obtained to verify design configuration and performance through off-limit and variability studies. These data also provide early reliability information to aid in the development and design of qualification and reliability tests, eliminating unnecessary testing and determining the need for specific tests.

APPLICATION APPROVAL TESTS

Application approval tests form the principal phase in the Confidence Development Plan. This phase, divided into qualification and reliability tests, provides for testing of parts, components, and systems at suppliers' and S&ID facilities. Successful completion of these tests provides the level of confidence necessary for application approval.

QUALIFICATION TESTS

Part and component qualification tests are conducted primarily by suppliers. System level tests are conducted at S&ID unless an entire system is procured from a supplier. These tests, based on vehicle specification criteria, form the basis for qualification approval, initially for battleship test, static firing, and finally flight hardware usage.

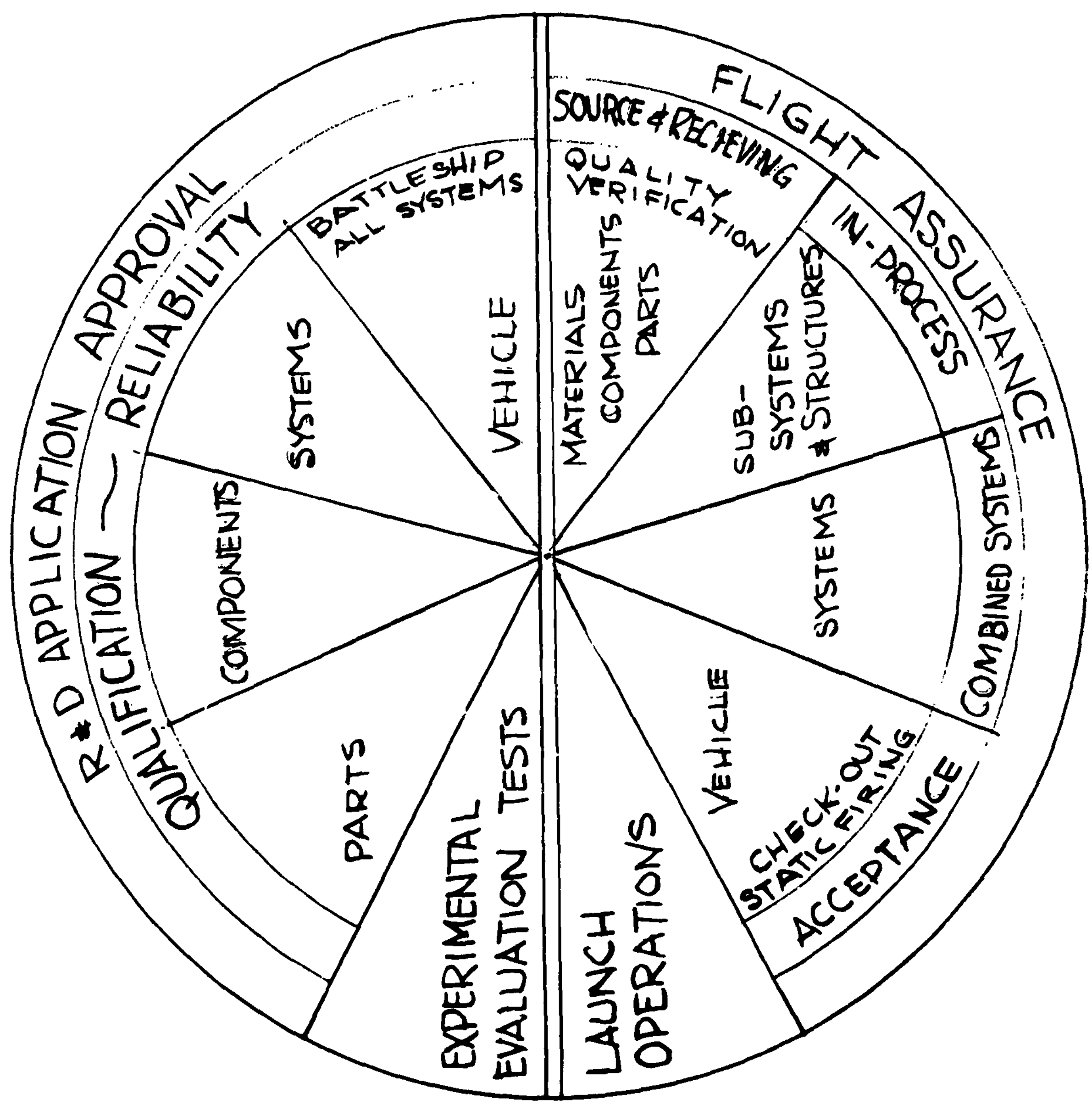


Figure 82. Confidence Development Plan

The component tests consist of designed experiments providing environmental combinations that simulate the combinations and sequences as they will appear in service. Functional parameter data is continuously recorded and supplied throughout the test program for immediate evaluation and utilization in design and reliability review of each item. System qualification tests are performed in accordance with the system specification, primarily to evaluate and qualify the interface functional parameters of the components within the system and the system output parameters. The tests are performed first under ambient conditions and later at selected environmental stress levels.

RELIABILITY TESTS

Reliability tests are an extension of qualification tests to provide additional data to uncover obscure failure modes and verify that probability of success and mean time-to-failure requirements are met. These tests complete the designed experiments and include, as a part of these experiments, the ultimate stress test to failure. This will provide data on failure modes and possible design improvement as well as delineation of failure distribution. Test articles previously used in qualification testing, as well as additional items, are used for these tests, subject to customer. These tests also provide quality verification.

ASSURANCE TESTING

The second major phase of confidence testing is that conducted on all flight hardware—from the component through the vehicle level, from the supplier's plant through S&ID Manufacturing, and including static firing and acceptance. These tests provide maximum assurance that each vehicle is rated for flight. They include functional cycling, with environmental exposure, and permit combined system evaluation before the vehicle is delivered.

SOURCE AND RECEIVING TESTS

Parts, material, and components are required at delivery to meet assurance tests under environmental exposures to verify that mechanical properties, organic compound properties, and performance parameters are within specified control limits.

IN-PROCESS TESTING

In-process testing is a continuous monitoring through all assembly operations including final vehicle checkout. All received hardware is tested during the assembly operation. Installed components and systems are functionally checked at the component, system, and combined system levels to assure that performance parameters remain within required limits.

COMBINED SYSTEMS CHECKOUT

Final combined systems checkout is primarily a thorough functional checkout to verify that functional parameters have not been affected by the assembly process and the vehicle system performance maintains a "GO" status.

STATIC FIRING

After completion of the combined systems checkout, the vehicle will be subjected to a series of static firings to obtain final assurance that the vehicle is ready for flight.

LAUNCH

Launch verification is the final phase of the Confidence Development Plan and the final proof of reliability. The telemetered launch and flight data of the vehicles flown will be evaluated and compared with the data obtained in previous ground testing. The vehicle launches will be verification and final demonstration of capability.

ECONOMICS AND OPERATIONS

SUMMARY

The study configurations described in this report were analyzed as to research and development, production, and operational costs. These costs were reduced to a cost per pound of payload in orbit as a basis for comparing the various configurations within each of the thrust classes and range of launch rates investigated.

In evaluating selected two-stage booster systems, there is no obvious cost optimum system within each of the 0.6, 0.8, 1.5 and 3.0 million-pound thrust classes. The systems utilizing existing state-of-the-art engines remain competitive with systems using advanced high-pressure engines due to the reduction in R & D costs and higher initial reliability. If the full R & D costs for the state-of-the-art propulsion systems were included, the configurations utilizing advanced high-pressure engines would show a distinct advantage.

The two-stage configurations that do show economic advantages are those that utilize clustered modules in the first stage and a single-tank second stage. If the two-stage boosters utilize common modules in both stages with advanced high-pressure propulsion systems, these systems are shown have the cheapest cost per pound of payload. These costs are considerably more pronounced with advanced high-pressure engines than with conventional bell-nozzle engines.

The cost per pound of payload decreases with increasing total thrust level for any given engine and propellant combination, with the greatest relative decrease occurring at the lower thrust levels of 0.6 and 0.8 million pounds of thrust.

The recoverable configurations within each thrust level studied represent the most economical method of orbiting payloads at launch rates of approximately 100 and above over a 10-year period. Below this launch rate the addition of a recovery system is uneconomical, primarily due to the added system costs and payload penalty associated with the recovery system.

The laterally staged booster configurations are limited in their payload capability but are economically competitive with conventional two-stage booster systems. This advantage is primarily at the lower launch rates of 50 per 10-year period and cannot be realized at higher total launches.

METHODOLOGY

Costs for a complete booster development and operational program have been estimated for each of the configurations in the study. These costs cover a 3-year development and 10-year operational launch period and have been reduced to a booster cost per pound of payload as a basis for configuration comparison. The operational launches occur in the 1965-1975 period.

The costs are compared both as to the minimum total program costs and the minimum cost per pound of payload. These cost parameters were studied over the prescribed launch rates to determine any cross-over or inflection points. The meaning of such points were examined as to their effect on ultimate vehicle selection.

The estimates were made by a standard North American Aviation costing procedure, supported by computer calculations, and the "Weapon System Cost Methodology" of the RAND Corporation (RAND Report No. R-287, 1 February 1956 and Revisions).

GROUND RULES

All costs are Fiscal Year 1962 dollars. No attempt has been made to estimate the effects of future inflation or deflation upon these costs. The labor rates, burden rates, and fees are those expected in Fiscal Year 1962 not in subsequent or post time periods.

Cost comparisons were made on a differential basis for comparison of concepts rather than absolute values. In this approach, consistency of methodology coupled with the selection of the most significant cost items results in data which indicate the relative economic position of each alternative. A tabulation was made of the cost categories which are most affected by the differences of the alternatives. The contents of each category were defined to permit evaluation and comparison with other alternatives.

When vendor supplied costs are used, a materials procurement cost has been added to their FOB estimates. All such costs are assumed to be area prices for budget and planning purposes only and are not firm quotations. The total program costs are believed to be within ±10 percent of what they would be for a firm quotation for the items included in each category.

No costs are included for the transportation of the vehicles from the manufacturer's plant to the launch site. These plants are assumed to generally be at dockside locations to facilitate water transportation where feasible. Real estate costs have been considered only for the manufacturing facilities costs. If new launching areas are acquired for this program, the costs of such land should be added.

The effects of variations in the recoverability rate upon the program costs were examined for each mission reliability selected.

APPROACH

The economic analysis of the study configurations were directed toward arriving at a booster dollars per pound of payload as a common comparison for each configuration. The expected results were to determine the least expensive or cost optimized configuration in terms of total program cost per pound of payload accelerated to a 300-nautical-mile circular orbital speed. The total program cost components of each configuration, consisting of R & D, production, and operations costs, were considered on the basis of thrust level for the four configuration groups — 0.6, 0.8, 1.5, and 3.0 million pounds of thrust.

It should be recognized that in an economic evaluation such as this where total program costs are considered, the magnitude of the fixed program costs tend to make the cost differential between one vehicle and the next relatively small. Furthermore, if the cost of the payload is added to the launch-vehicle program costs, the percentage magnitude of the cost differential between programs becomes much smaller. These observations are particularly true if reliability and payload capability are not included in the program evaluation. These two items provide the major cost differential between otherwise similar programs. Difference in payload capability affects the dollars-per-pound-of-payload comparison, although total program costs are equal. Similarly, reliability affects the dollars per-pound-of-payload comparison, although total program costs again may be equal. Two vehicles may have equal total costs, but the vehicle which carries the most payload to the desired mission would have the smaller program costs, on the basis of dollars per pound of payload, with equal vehicle reliability. These two programs may involve one vehicle with a 90 percent average operational reliability and another with an 80 percent average operational reliability. Obviously, amortizing the losses due to the lack of reliability against the total program price would make the less reliable vehicle 10 percent more expensive in terms of dollars per pound-of payload.

Tabulated cost results will sometimes reveal the lowest cost-per-pound-of-payload vehicles as a function of the total number of launches in

a 10-year operating period, but it should be noted that when these results are plotted versus total payload, a change of conclusions as to the most economical systems may occur. Since it is assumed that the purpose of the program is to launch hardware and personnel, the plots of cost per pound of payload versus successful payload is a more significant comparison.

PROGRAM SCHEDULE AND REQUIREMENTS

A development and phasing schedule has been established for the selected state-of-the-art two-stage 3-million-pound thrust booster system. The development schedule for this selected typical booster system using two F-1 engines on the first stage and a single J-2 engine on the second stage is presented in Figure 83. All key phasing dates, program milestones, manufacturing rates, hardware quantities, test rates, launch rates, and operational quantities are shown in this schedule. The interphase relationships of the various major program items also are shown as well as the lead times necessary to meet scheduled release or acceptance dates.

The master development schedule covers the major phases necessary to support two operational concepts of attaining 10 and 100 launches per year within a 10-year period. The other operational concept of one launch per year within the 10-year period was not scheduled because its phasing effect on the program could easily be interpolated from the other two rates. The cost implications of the various launch rates are covered in the cost trade-off section.

The length and phasing of the program plan for this typical configuration are determined by the availability of the launch facilities for the first operational launch. This is the longest lead time item of the program. Crash funding should be applied to provide these facilities at an early date. Both engines are in development and are estimated by the engine suppliers to be available in early 1963. Development of the booster vehicle tanks and internal systems is phased to provide a complete booster system at the time the propulsion subsystems are available for this program. This would provide for the first operational launch, including all necessary vehicles and ground support equipment, by December 1964, 36 months after go-ahead.

The engineering efforts and time spans for design and development of the booster tanks and systems, other than the propulsion systems, are the same for all configurations in this study. For all vehicle configurations using other than F-1 or J-2 engines, the length and phasing of the program are determined by the availability and delivery of the first flight engines. These engines are not yet in development but the suppliers estimate that they will be available between 36 and 48 months from funded go-ahead.

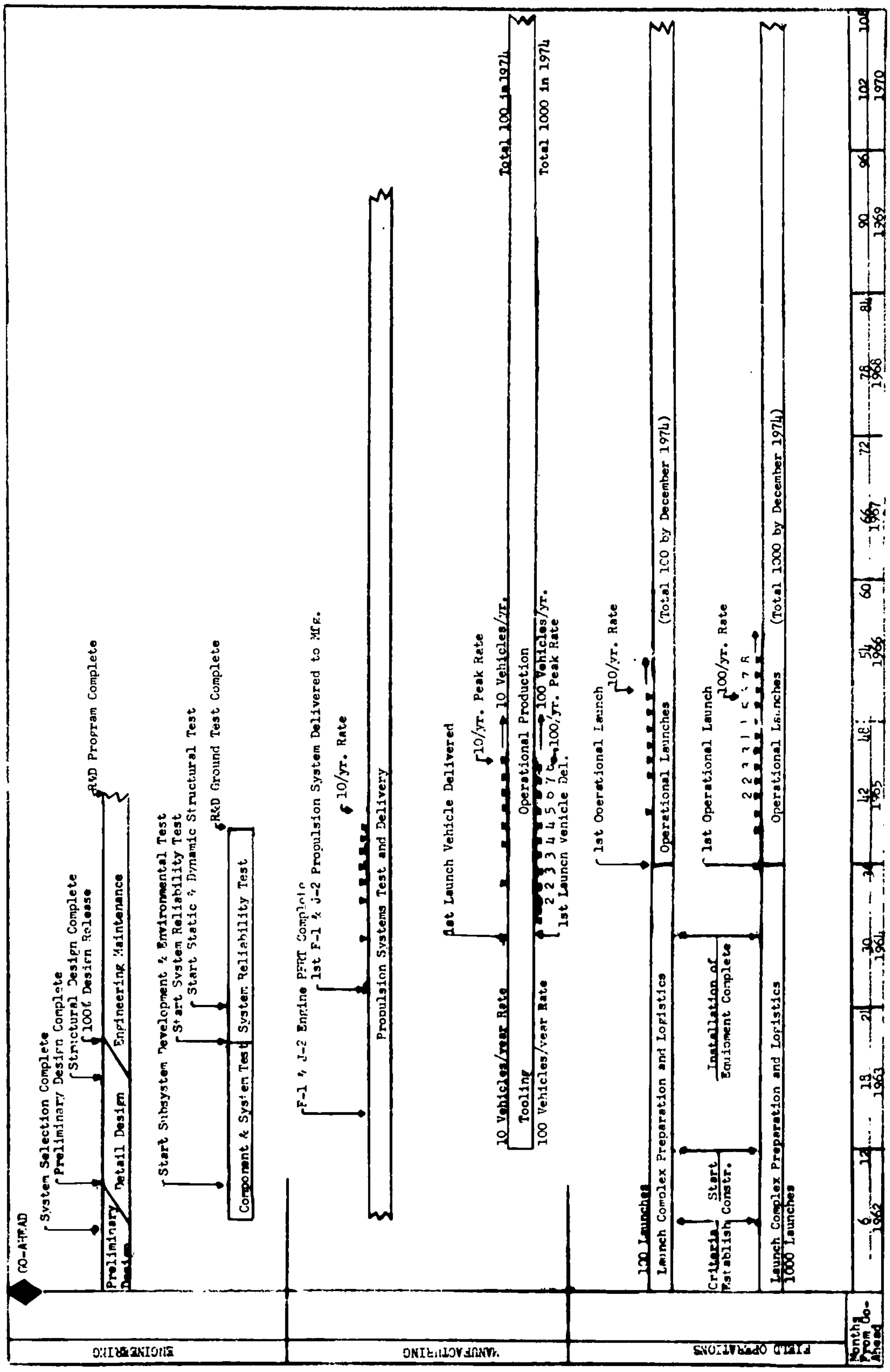


Figure 83. Development Plan for Representative Booster Configuration

Review of the various techniques covered in the study indicates that there would be no significant difference or increase in development time for the recoverable over nonrecoverable configurations, because the longest lead time items for the booster development are the launch complex and delivery of advanced-type propulsion systems. Economic comparisons made between expendable and recoverable systems indicate that the paraglider system promises to be the lowest-cost recovery method. Development of the paraglider recovery system would be scheduled for operational launches in late 1964. Paraglider recoveries of the Saturn S-I (C-2) systems are currently planned for the same period. Recovery of the Saturn C-1 system could be made as early as October 1963.

Engineering

At the start of the R & D program for a vehicle in the 0.6-to-3-million-pound thrust range, basic system research would be conducted in the areas of propulsion, guidance and control, high-strength materials, and anticipated problem areas to achieve the following objectives: (1) to select the most advanced subsystems and components, (2) to establish detail requirements for the vehicle, and (3) to permit a reasonable degree of engine-vehicle optimization. The development plan depends upon parallel development of the booster tanks, subsystems, and engines. Actual development of the F-1 and J-2 engines has been in progress at least a year in advance of the required booster fuel-tank development start date. This is necessary because of the long lead time requirements of the propulsion systems.

The program plan and the high reliability goals of 90 to 95 percent provide for design of the vehicle in the operational configuration from the outset, with special kits and modifications preplanned as necessary to conduct the ground-testing and development program. The anticipated reliability level at the end of the ground-testing program eliminates the necessity for R & D launches and provides for the first flight to be a full operational launch. The size of the vehicle requires that logistic concepts be incorporated immediately in the preliminary engineering design considerations.

The following program plan description is typical for high-thrust boosters in the 0.6 to 3-million-pound range and is based on the developmental configurations which use primarily F-1 liquid oxygen and RP-1, and J-2 liquid oxygen and hydrogen engines. This program provides the earliest launch date with nominal funding. Increased or crash funding on the liquid oxygen and hydrogen engines for the other developmental configurations could possibly provide the same program dates.

The actual design engineering phase for the booster tanks and systems, assumed to start about 1 January 1962, would require about 21 months for 100-percent of design release. Structural release of the airframe would precede the 100-percent design release by 3 months. Design engineering maintenance would start after design release and continue for the duration of the launch program. The initial design would provide for later increases in production rates, removing R & D provisions and incorporating design refinements during the ground tests and the later operational launch program.

Basic problems would be solved by analytical and research testing techniques during the preliminary design and early detail engineering phases. Research would provide information necessary to effect the booster concept before the larger expenditures required for hardware are committed.

During the preliminary design period, an architectural and engineering contractor would be selected and preliminary design of an operational launch complex and recovery base would be started. Since the construction of the launch complex is the longest lead time item in the development schedule, the design and construction of the complex should be started as soon as the facilities criteria are established. By the time preliminary design is completed, a full-scale structural mockup will have been constructed for use in determining details of installing internal systems and components. Selection of all major systems would be made during the preliminary engineering phase.

Continuous refinements would be accomplished throughout the development and ground test program to provide booster vehicles with high reliability. Additional aerothermodynamic studies and wind tunnel tests would be conducted to support the booster design and substantiate ground test data. The vehicles and systems would be tested under operational environmental conditions. Spacing between tests would provide time for data feedback and analysis of booster system performance.

A comprehensive reliability program would be established for the booster system to assure attainment of the performance reliability goal of greater than 50 percent when the vehicle becomes operational. This goal can be achieved by a vigorous coordinated reliability program which would have as an objective a reliability value of at least 80 percent at the first booster launch. This will eliminate the necessity for R & D launches. Reliability must be emphasized throughout the development program and would consist of a combination of quantitative analyses, coordinated vendor and supplier practices, and an integrated test program.

Ground Testing

Development and laboratory testing of booster systems and components would follow a program parallel to the design engineering phase. Various materials, processes, and structural components would be tested to determine suitability for specific applications. All individual systems would be extensively tested for design and reliability evaluation and to increase reliability prior to actual operational launch.

After acceptance from manufacturing, an airframe would undergo static, dynamic, and fatigue testing to determine performance under all conditions anticipated in operational use. Sufficient ground tests would be conducted on the systems to clearly demonstrate reliable combined systems operation. The ground test program would include testing of checkout equipment operation, laboratory simulation of launch operations, evaluation of operational concepts and procedures, investigation of malfunctions, and determination of service life of the system.

Manufacturing

Tooling and manufacturing would include operational vehicle production and production of ground support equipment. Production rates to support the operational concepts of 10 and 100 launches per year over a 10-year period are shown in Figure 83. Tooling development would begin approximately 18 months prior to the acceptance of the first launch vehicle. Basic tooling for the large booster vehicle manufacturing program would probably be keyed to the operational production requirements of 100 vehicles per year. The lower rate of 10 per year could be provided for by leaving the booster structure in the major assembly jigs for a longer time and by lower manpower loads. This flexibility in the manufacturing rates would avoid major changes in the tooling setup if the development program should change from manufacturing 10 per year to 100 per year. Tooling for the 10-per-year launch rato would be minimized.

Most of the manufacturing would be performed in the major assembly jigs with very little support tooling to provide for manufacturing on an individual, rather than a production basis. Production methods would be involved at production rates of 100 per year or more.

R & D Fabrication

A static airframe would be manufactured to support the structural, dynamic, and fatigue testing program of the laboratory. Structural and functional tank segments would be provided to conduct laboratory functional tests, such as stage separation and structural joint tests. Critical

components and subsystems and one set of ground support equipment would be manufactured for use in an integrated R & D ground test program which would include reliability and combined system testing.

Operational Production

To meet an operational launch date in December 1964, a hardware production go-ahead would be required by early 1963. This scheduling would provide sufficient boosters to meet the operational launch requirements starting in late 1964. Production build-ups have been established to support the yearly launch rates of 1, 10, and 100. The 1-per-year production rate would provide 10 vehicles for the 10-year operational period, the 10-per-year rate would provide 100 vehicles, and the 100-per-year rate would provide 1000 vehicles.

The 10-per-year launch rate would require three sets of ground support equipment. Two of these would be new and one would be refurbished from the R & D program. The 100-per-year launch rate would require nine sets of ground support equipment and one refurbished from the R & D program. One set would be used at each of the launch preparation stations and pads, and one at the maintenance support area.

Facilities

R & D ground test and operational launch requirements, as well as criteria, must be established early in the engineering program to provide the lead time necessary for construction of launch complexes and other basing facilities. Design and development of the R & D ground test facilities and operational launch complex facilities would be started in early 1962 before completion of the preliminary vehicle design. Construction of the R & D facilities would be completed prior to the first vehicle acceptance from manufacturing in 1964. Design, construction, and equipage of the operational launch complex would be completed in time for the first operational launch in December 1964.

Manufacturing Requirements

Sufficient development facilities exist in a number of aircraft and manufacturing plants to provide the required engineering, laboratory testing, and detail manufacturing. However, final launch preparation and checkout operations for the operational launch program should be conducted near the launch site or water transportation because of handling problems with large vehicles. Engineering, logistics, and laboratory testing for the booster development would require about 200,000 square feet of facilities, while general supporting activities would require about 400,000 square feet.

Manufacturing facilities required for factory-site detail fabrication would involve approximately 150,000 square feet of space foı the 1-per-year rate, 350,000 for the 10-per-year rate, and 500,000 for the 100-per-year rate.

Operational Launch Requirements

The operational launch complex is scheduled for occupancy 6 months prior to start of operational launchings in 1964. At this time, crews can receive and check out the vehicles and ground support equipment as it is installed in the launch areas. Alignment and operation of all checkout, handling, and downrange instrumentation would be conducted during this period.

It is estimated that two operational launch pads would be required to conduct the 10-per-year launch rate, and that eight would be required for the 100-per-year launch rate. Launch preparation time for each vehicle will require an average of 2 months during the 10-year operational period. This time span would probably be longer in the early part of the program with later operational requirements establishing shorter time on the pad. These shorter periods, with the attendant reduction in facilities requirements, might be realized soon after the peak launch rate is achieved. Each of the launch pad requirements for the yearly launch rates include one pad as back-up in the event of disastrous malfunction of a launch vehicle. The back-up pad could be used either in rotation with the other pads or exclusively for static firing tests until required for launching.

Final Preparation and Launch

Because of the extreme size of the systems, the booster stages would be mated and checked out at a preparation building located on the launch site. The actual launch operations would be conducted in an area removed from the launch preparation building because of detrimental environmental effects and potential hazards, such as high acoustic levels, blast, propellant and exhaust toxicity, and other range safety considerations.

Logistics Support

A logistic support program would be established to insure the avaliability of crews, spares, ground support equipment, propellants, and other requirements for the booster vehicle program, and to aid in the development of hardware with inherent maintainability characteristics. Mission and operational goals would establish requirements for ground support equipment, maintenance level and supply, depot level support, maintenance facilities, manpower, spares, and maintainability characteristics in design.

Advantages are felt to exist in locating the preparation, test, and maintenance facilities adjacent to the launch site, as well as providing on-site manufacturing capability for the cryogenic propellants and large capacity storage for the noncryogenics. An alternative to on-site capability is the location of all facilities adjacent to navigable waterways with water transportation available.

Contractor training effort would begin shortly after 100 percent design release in September 1963, and would consist of formulating the training plan, preparing manuals and equipment, and conducting the courses required for the operational launch crews. During the formulation of the training plan, the vehicle system and the plans for its operation would be analyzed in detail. Results of this analysis would establish qualitative and quantitative system manning for the booster vehicles, including requirements for training and training equipment development.

IBM MECHANIZATION

A comprehensive computing machine program covering performance and weights analysis was extended to include the calculation of production costs. Certain quantities which were obtained from the weights program were readily useful in the cost calculations. Among these were the AMPR weight, the type and quantity of engines, the propellant weight, and the payload weight. From the vehicle description and AMPR weight, the fabrication and tooling costs were estimated; from the fuel and oxidizer description and weight, the propellant costs were estimated. In a similar manner the engines were costed, using vendor data where available. A calculation procedure was adopted which permitted the evaluation of both modular and recoverable alternatives as well as the conventional single-tank designs. By use of this procedure, the lowest production cost per pound of payload could be readily obtained and compared for each configuration and the most promising determined. It was not assumed that the highest payload configuration would necessarily be the most attractive. Estimates of engineering, maintenance, launch facilities, and operations costs were made to compare different configurations. However, these were considered a function of thrust level and complexity and were not included.

In algebraic terms, the following quantities were given:

A=AMPR weight per module

w_p=Propellant weight per module

Let

N=Number of launches in the program

n_m=Number of modules per stage (n_m = 1 for single tanks)

n_e=Number of engines for stage

C_a=Fabrication cost per pound of AMPR weight of total Nn_m for the vehicle

C_T=Tooling cost per pound of AMPR weight

C_e=Unit cost of the engine at the total Nn_m for the vehicle

K=Number of sets of tooling required by the total Nn_m for the vehicle

From the above, the following costs per stage are readily calculable:

1. Fabrication $= Nn_m C_a A$
2. Tooling $= C_T A\ K$
3. Engine $= Nn_e C_e$
4. Propellant $= Nn_m W_p C_p$

Actually, the total production cost for any stage equals the sum of the above, less the fuel. The fuel cost was added because it is so readily determinable from the weights and performance programs. The total cost is then equal to the sum of the above terms for each stage.

$$\text{TOTAL COST} = \sum_{S=I,\,II} (Nn_m W_p C_p + Nn_m C_a A + C_T A\ K + Nn_e C_e)$$

Where the same modules or engines are used in both stages, the sum per vehicle must be used to determine the total cost. The total payload successfully placed in orbit is equal to the product of the average mission reliability and the total theoretical payload launched and is equal to p N W_{pl}. In this term, p is the average probability of successfully placing the payload in orbit. Thus, the total production cost per pound of payload successfully placed in orbit is defined as follows:

$$\frac{\sum_{S=I,\,II} (Nn_m W_p C_p + Nn_m C_a A + C_T AK + Nn_e C_e)}{P\ NW_{pl}}$$

If either modules or engines are common to both stages in the above equation, the cumulative average unit cost must be determined based on the total quantity.

$$\sum_{S=I,II} Nn_m \text{ or } \sum_{S=I,II} Nn_e o$$

The total cost can then be determined by multiplying the unit cost by these quantities.

In the event that the first stage is to be recovered, an estimate of how many boosters will be required must be made. The term p_r is called the "recoverability probability", but is actually the product of the stage mission and recovery reliabilities. It can be demonstrated that the number of times one booster can be used is equal to 1/(1-pr), or R. Thus, a new generalized total cost relationship can be obtained as follows:

$$\text{Total Cost} = \frac{\sum_{S=I,II} \left[Nn_m W_p C_p + (\frac{N}{R}) n_m C_a A + C_T AK + (\frac{N}{R}) n_e C_a \right]}{p\ NW_{pl}}$$

The quantity N/R is the number of boosters required to perform N launches. For example, if it is required to make a total of 100 launches using boosters whose average recovery reliability is 0.87 for 100 launches, the number of uses which one might expect is 1/1-0.87 = 7.70 per booster. Thus, the number of boosters required would be 100/7.70 or 13 units. The booster fabrication, tooling, and engine requirements would be based on this number of units. The effect of tooling costs would be to reduce the number of sets of tooling and tooling maintenance required.

The possibility of reducing the arithmetic portions of the hand calculations by use of computing machine calculations for research and development, engine development, ground support equipment, facilities and operations costs was considered, but was not used as a considerable quantity of itemized calculations would be necessary to determine the relationship of cost to thrust and launch rate. The determination of all of these relationships was not necessary for the initial selection optimum configurations. Consequently, machine calculations used did not include these factors. They may, however, be included in future studies of a similar nature.

TWO-STAGE BOOSTER SYSTEM EVALUATION

The first phase of the study covered primarily selected two-stage vehicles in the lower thrust range. This part of the study shows the desirability and feasibility of developing two-stage vehicles for high-payload orbital missions by 1965. Boosters with low dollars-per-pound-of-payload costs can be available within the next 3 years using combinations of existing state-of-the-art J-2 and F-1 engines in the first and second stages or within 4 years if advanced high-pressure engines are used in both first and second stages. The second phase of the study covered comparison of alternate systems with the selected two-stage systems.

CONFIGURATION COMPARISONS AND TRENDS

The most promising two-stage vehicles selected for further analysis in the study fall into two booster configuration groups, (1) boosters using state-of-the-art engines such as J-2's and F-1's and (2) boosters using advanced high-pressure LO_2LH_2 engines.

The trade-off curves showing dollars per pound of payload versus launch rates for the four thrust levels of 0.6, 0.8, 1.5, and 3.0 million pounds of thrust are given in Figures 84, 85, and 86. The figures show that the cost per pound of payload decreases with increasing total thrust level for any given engine and propellant combination with the greatest relative decrease occurring at the lower thrust levels of 0.6 and 0.8 million pounds of thrust. In comparing the state-of-the-art systems with the advanced high-pressure systems, it can be noted that there is no significant difference between the two within each of thrust classes and launch rates covered. The systems utilizing existing state-of-the-art systems remain competitive with systems utilizing advanced high-pressure engines due to the reduction in R & D costs, higher initial reliability, and lower payloads. If the full R & D costs for the state-of-the-art propulsion system were included, the configurations utilizing advanced high-pressure engines would show a distinct advantage.

RESEARCH AND DEVELOPMENT COSTS

Engineering Design and Development

Engineering design and development costs were obtained by an analysis of the manpower and laboratory support required to design and develop the

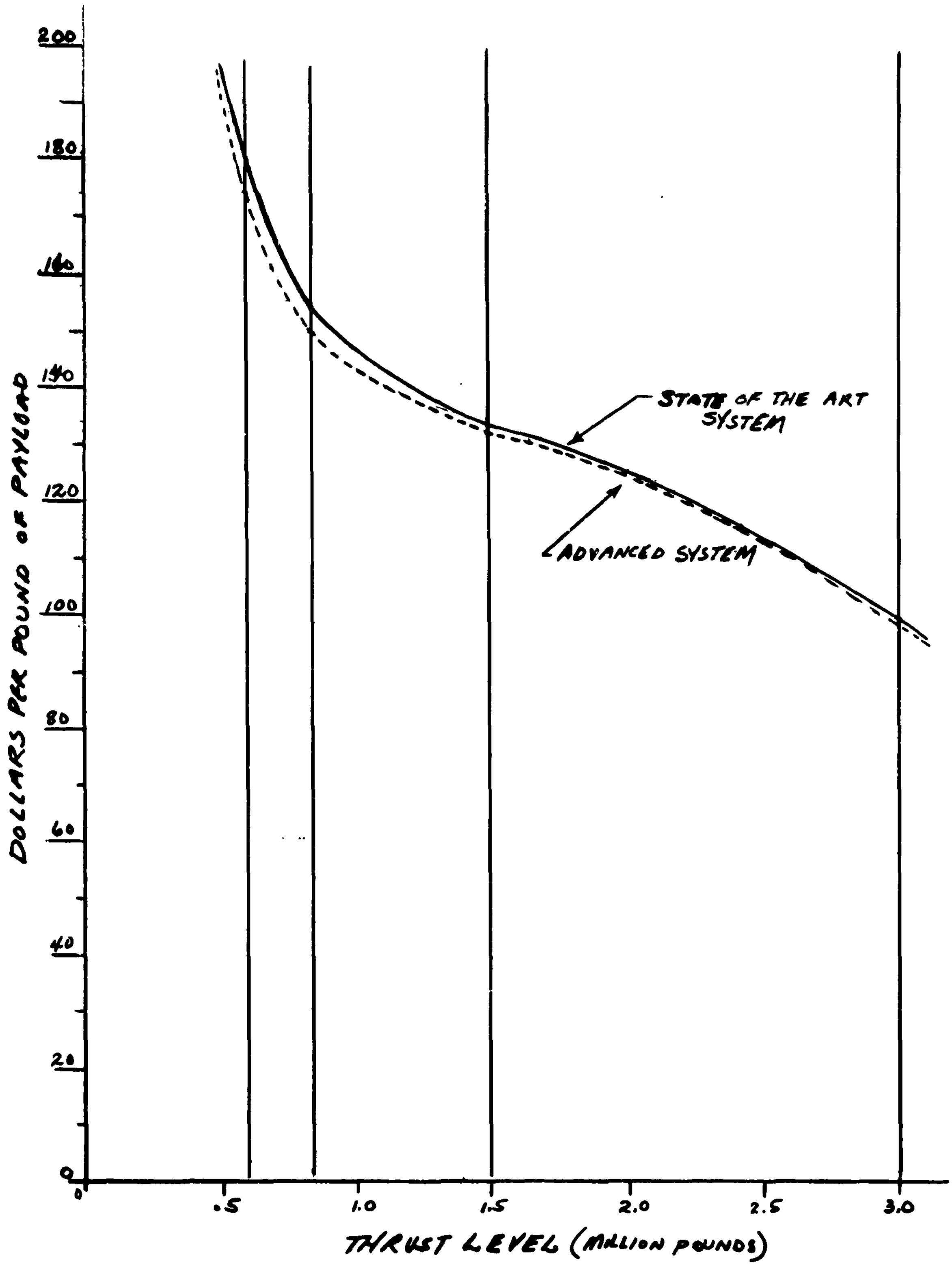


Figure 84. Selected Two-Stage System Cost Comparison at 100 Launches

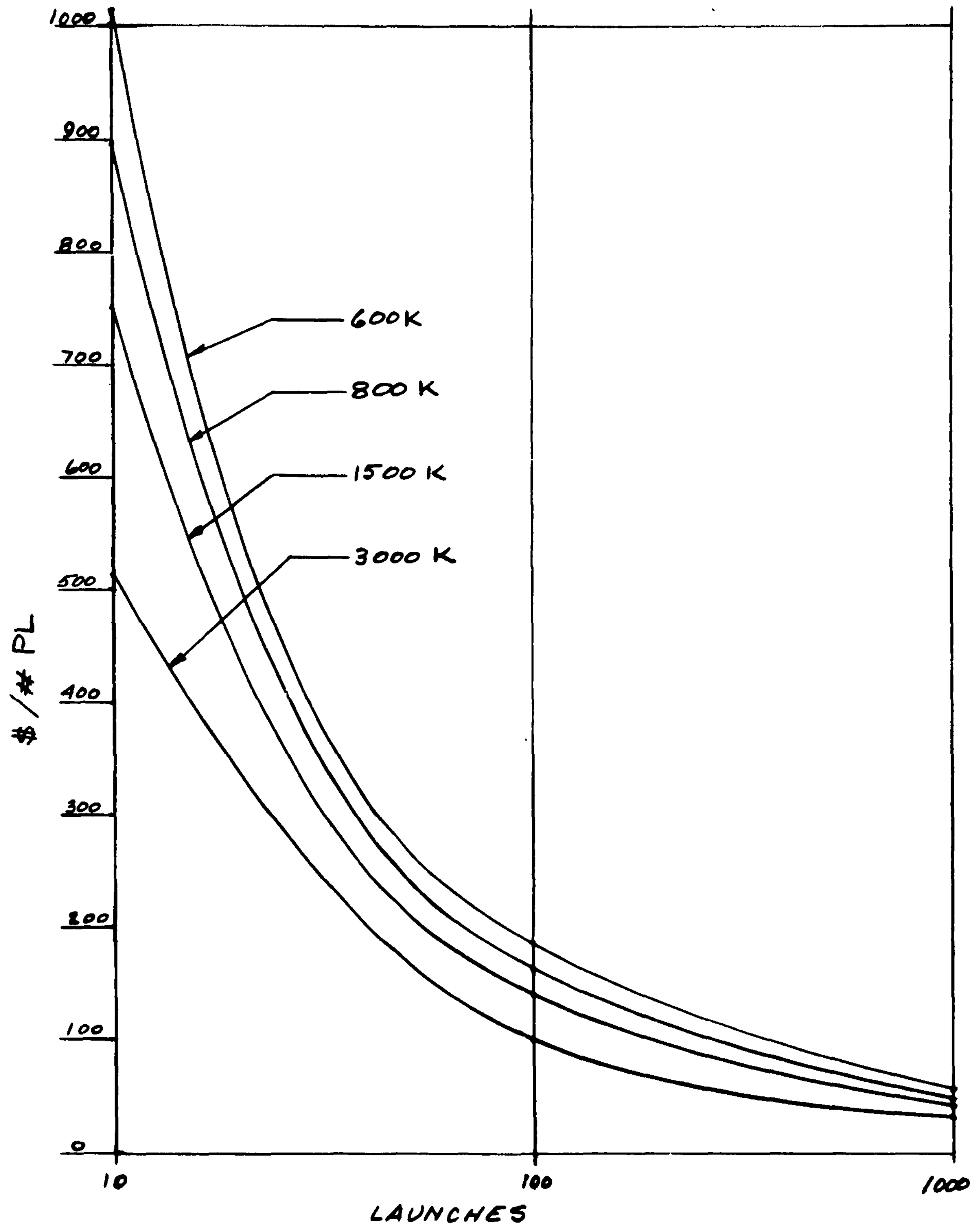


Figure 85. Selected Two-Stage State-of-the Art Systems Cost Comparison

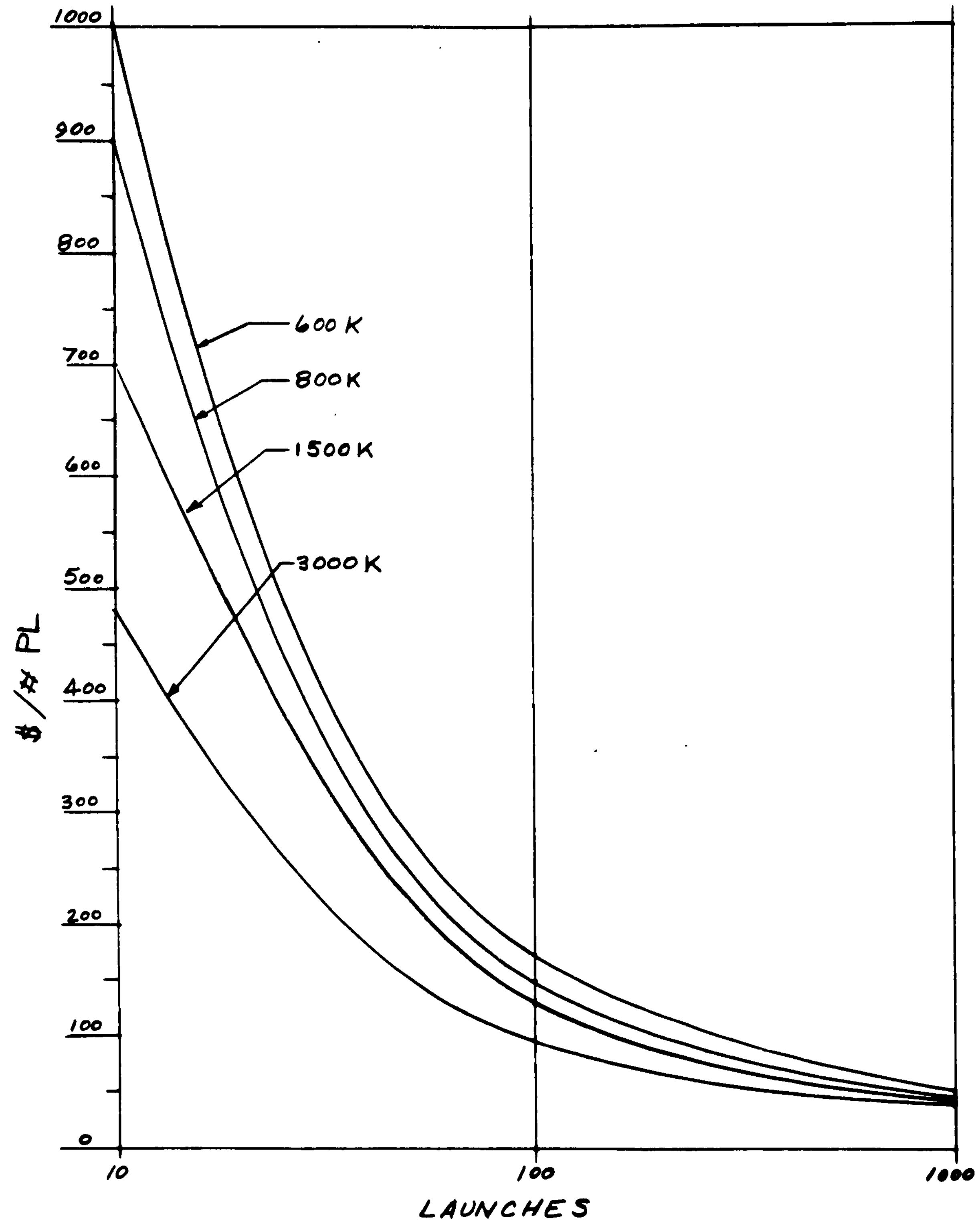


Figure 86. Selected Two-Stage Advanced Systems Cost Comparison

various configurations. All preliminary and detailed design, laboratory support, and production engineering costs have been included. These costs represent the requirements to accelerate attainment of an operational reliability of greater than 50 percent for each configuration at the first flight. Shop support, technical data and publications, and associated items are also considered in these costs. The preliminary design portion includes costs for aerothermodynamic design and analysis, booster, ground support equipment design and project management costs. The detailed design is comprised of aerothermodynamics, airframe, secondary power, propellant systems, separation, flight control, radio command guidance system, propulsion integration, instrumentation, and ground support equipment. Support items are included for project management, systems integration, reliability, human factors, technical data and publications, and production engineering.

Engine Development

The R & D costs for the new LO_2/LH_2 and LF_2/LO_2 engines and the solid propellant motors were obtained by vendor estimates based on an assumed development program. In the solid-propellant stage development program, it was assumed that 10 full-scale motors and 20 subscale motors will be fabricated and fired. These together with the tooling and equipment required for their fabrication, transportation, and testing, will comprise the bulk of the development costs. No firing of a seven-motor cluster is included in these estimates because the individual motor tests are believed to be sufficient. No R & D costs were included for the J-2 and F-1 engines since this has been previously funded. Where modification of these existing engines was necessary a modification cost was included.

The development programs of the LO_2/LH_2 and LF_2/LO_2 engines, from which cost estimates were made, assumed that no parts, devices, or production techniques used in the F-1 program are applicable to the new engine. This assumption places the liquid and the solid propellant costs on a similarly conservative basis.

Ground Support Equipment

The cost of all ground support equipment chargeable to the booster system has been included in this item. All implements or devices required to inspect, test, adjust, calibrate, transport at the launch site, service, launch, and otherwise support and maintain the functional operating status of the booster system, subsystems, and components are considered. However, the facilities or equipment required to transport the boosters from the manufacturing site to the launch area are omitted. Such transportation might be accomplished by ship or by a special transport system if the

manufacturing facility is relatively near the launch site. It should be noted that such transportation costs have been omitted throughout the study.

Recovery System Development

The development costs of the paraglider recovery system were determined on the basis that the first flight of the booster vehicle will be operational regardless of the development status of the recovery system. The paraglider on the Saturn S-I (C-1) vehicle will be in process of full-scale testing at the time of this launch.

PRODUCTION COSTS

Manufacturing Facilities

The manufacturing facilities estimates have been based upon an average production of 100 vehicles in a 10-year period on a single-shift basis, and the production of 1000 vehicles on a two-shift basis. The facility requirements for a total production of 100 vehicles over a 10-year period would be the same as for the 1000 but would require only one-shift production.

All details are to be manufactured off-site. Covered building area provides for subassembly, final assembly, mating of tank and engines, warehousing, minimum tooling, minimum office area (shop supervision only), first aid, an X-ray laboratory, a minimum maintenance area, aisles, and utilities in excess of those presently existing in the aircraft and manufacturing industry.

Basic Tooling

Tooling costs have been calculated after consideration of the requirements of each configuration covered in this study. This cost includes the design and fabrication of the tooling required by the assumed production rates. Contractor supplementary tooling costs have been included. The latter item includes all tooling jigs, fixtures, and devices which are not expended in the actual production process. Masters, gages, and checkout equipment used on the production line are included in the supplementary tooling costs. In general, the basic tooling cost includes all machine tools, hand tools, welding equipment, gages, quality control devices, heat treatment equipment, forming tools, insulation application machinery, equipage of a meter room, materials laboratory equipment, production jigs and fixtures, and hydrostatic test equipment.

Vehicle Fabrication

Fabrication costs are based upon the use of aluminum semi-monocoque structures for all first-stage configurations of this study. The second stage requires overall insulation and ablative or other heat resistant material on all leading edges. These materials are attached to the outside of the basic aluminum semi-monocoque structures and increase the basic second-stage cost. The cost of integrating the tank and engine systems has been included under this category.

The costs are based upon calculations of labor, material, burden, and fee for both the single- and modular-component types. From these, a cost per pound of AMPR weight was obtained. This cost was determined for each stage of the configurations because the upper stages are more complex. The costs were then extended from one initial point on an 85 percent learning curve.

Engine Production

The engine production costs were obtained from Rocketdyne Division of North American Aviation for the F-1 and J-2 engines. These production costs were modified to allow for spares and procurement costs. Other engine costs were obtained as a composite cost from several vendor estimates.

Guidance and Flight Control Production

The guidance and flight control items include a radio command guidance system located in the second stage, and associated flight control items located in each stage. All equipment necessary for the successful performance of the vehicle in accelerating the payload to the desired velocity at the proper attitude has been included. Any guidance or flight control equipment required by the payload for the accomplishment of its mission, such as retro rockets, stable platforms, or autonavigators, has not been included.

Engineering Maintenance

The engineering effort required for the production buildup to support the assumed programs has been included. This quantity reflects the design changes and general shop liaison required for the 10-year operational period. During this time, the engineering maintenance effort equals or slightly exceeds the basic engineering effort itself. While there is actually no sharp cut-off in this function, it may be considered as all effort after 100-percent design release.

Tooling Maintenance

Tooling maintenance effort is required to continue the production rates required for the 10-year program. The basic tooling costs per set were extended on a 108-percent learning curve versus the total number of vehicles produced. The cost increase obtained by taking the difference between the cost of the number of vehicles produced and the basic tooling costs represents the amount necessary to maintain the production rates of the individual programs.

Propellant

The total cost required for two launches per vehicle was included. The additional load of propellant was allowed for static firings and boil-off of the volatile propellants. The costs used were obtained from the latest vendor quotations and do not reflect the cost of facilities required by the launch rates of this study. The propellant manufacturing facilities required have been included in the launch facilities item, with the exception of RP-1, which would be purchased from existing sources.

Paraglider Fabrication and Vehicle Refurbishment

Where a recovery system is used, the same costing approach is generally followed. In the case of the paraglider, costs for the sail, booms, cabling, and control devices have been included. In this case, it was assumed that the entire paraglider system is replaced for each launch. The other recovery systems were costed as part of the vehicle fabrication. An additional refurbishment charge is made on each recovered booster of 15 percent of the fabrication cost for the paraglider system only.

LAUNCH OPERATIONS

Launch Facility Costs

The launch facilities costs include those for the launch pads, fuel and communication lines, launch pedestals and supports, water deluge systems and flame deflectors, blockhouses and launch control centers and their equipage, launch shops, assembly and instrumentation buildings, pressurant supply systems, warehouses, launch pad roads, aprons, and combination access towers and gantry cranes. In this cost analysis, only the conventional, vertical on-pad assembly launch complexes have been considered. All of these have been estimated for the complete vehicles and does not include the existing facilities at AMR which may be available to support this booster program.

The cost of one launch-tracking and communication station at the launch site is also included. The costs include engineering administration, flight

control, data processing, tracking, and communications complete with their operating and support requirements.

All estimates assume that space is available and that the launch base will be constructed at AMR. No allowance has been made for remote launch facilities.

It is assumed that the tanks will be hydrostatically tested by the manufacturer before shipment and that tank and engine mating will be conducted at the manufacturing facility.

The base site cost consists of base administration, personnel and industrial relations, engineering, laboratory, and cafeteria and support facility buildings. Such additional support items as a fire station, a sewage disposal plant, and security facilities are included, together with their required equipage costs. All these are in addition to the existing facilities at AMR. The machinery and equipment, base machine shop, and storage sheds are also included as components of the total base support facilities.

Included is the cost of a 50,000 kilowatt base electrical supply complete with its distributor system, a 3,000 gallon per day water supply, a distribution and storage system, base communications, and associated intrabase highway facilities.

In estimating the size of the required fuel manufacturing and storage facilities, twice the amount of fuel required for the launch of the largest vehicles has been allowed for static firing. Static firings will be determined after an acceptable reliability level has been reached. Thereafter, the excess area previously used to manufacture and store fuel will be otherwise employed without interrupting the launch schedule.

The fuel manufacturing and storage facilities costs include those for manufacturing liquid oxygen and liquid hydrogen and storage tanks only for RP-1.

Launch Operations

This item includes the requirements for manpower, facility maintenance, support facilities, and logistics and training at the launch site for the 10-year operating period. It was assumed that the launch facilities require 10 percent per year maintenance. Thus, the maintenance over a 10-year period is equal to the original cost. The support facilities maintenance includes the base administration, equipage of base facilities, base maintenance and utilities, and transportation maintenance. The logistics and training at the launch site base include the charges for logistics engineering and field service representatives, training and train-aids, transportation of the booster, payload and spares on the base site, parts depot manning, and spare parts replacement labor.

Recovery Operations

An estimate made of the recovery operations costs included manpower, facility maintenance, and logistics items. The manpower item included direct and support salaries for the 10-year operating period. The maintenance of the launch base recovery facilities and the recovery site support facilities was also estimated. The logistics and training at the recovery site base included estimates of logistics engineering, field service representatives, depot manning and spare parts replacement labor, and nitrogen purging costs.

TOTAL COSTS

The ratios between the three major cost elements (R & D, production, and operations) vary with launch rate when considered on a cost per pound basis. The higher the launch rate, the more significant the production costs per pound of payload becomes. Figure 87 shows the total program cost breakdown for the 0.6 million pound thrust advanced system. At 10 launches per 10-year period, the R & D and operations costs are 85 percent of the total with production amounting to only 15 percent. At 100 launches per 10-year period, the R & D, production, and operations costs per pound of payload are all approximately equal. At the higher rate of 1000 launches for the same period, the production costs per pound of payload constitutes about 80 percent of the total costs. These ratios were typical for all the two-stage systems studied.

A breakdown of the total production costs based on varying launch rates is shown in Figure 88 for the same two-stage 0.6 million-pound thrust configuration. This curve illustrates that, for vehicles in the low thrust ranges, the engine costs are nearly 50 percent of the total production costs with the balance of the costs covering tank fabrication, tooling, and propellants.

When the costs for each configuration were tabulated, it was revealed that the higher the first-stage thrust level, the cheaper the dollars per pound of payload for any given launch rate (see Table 25 and Figures 85 and 86). However, it should be noted in Figure 89, that if the dollars per pound of payload orbited is compared on the basis of total successful pounds orbited rather than launch rate, the reverse is true. Figure 89 shows the lower the thrust level, the lower the dollars per pound of payload successfully orbited. This is a result of the smaller first-stage system orbiting payload earlier at a higher launch rate than the larger system, coupled with reduced prices through high production. If an extremely large amount of payload is orbited, the larger system will eventually show an advantage. Since the purpose of the mission is to place payload in orbit, it is more reasonable to compare costs of the same successful payload than at the same launch rates.

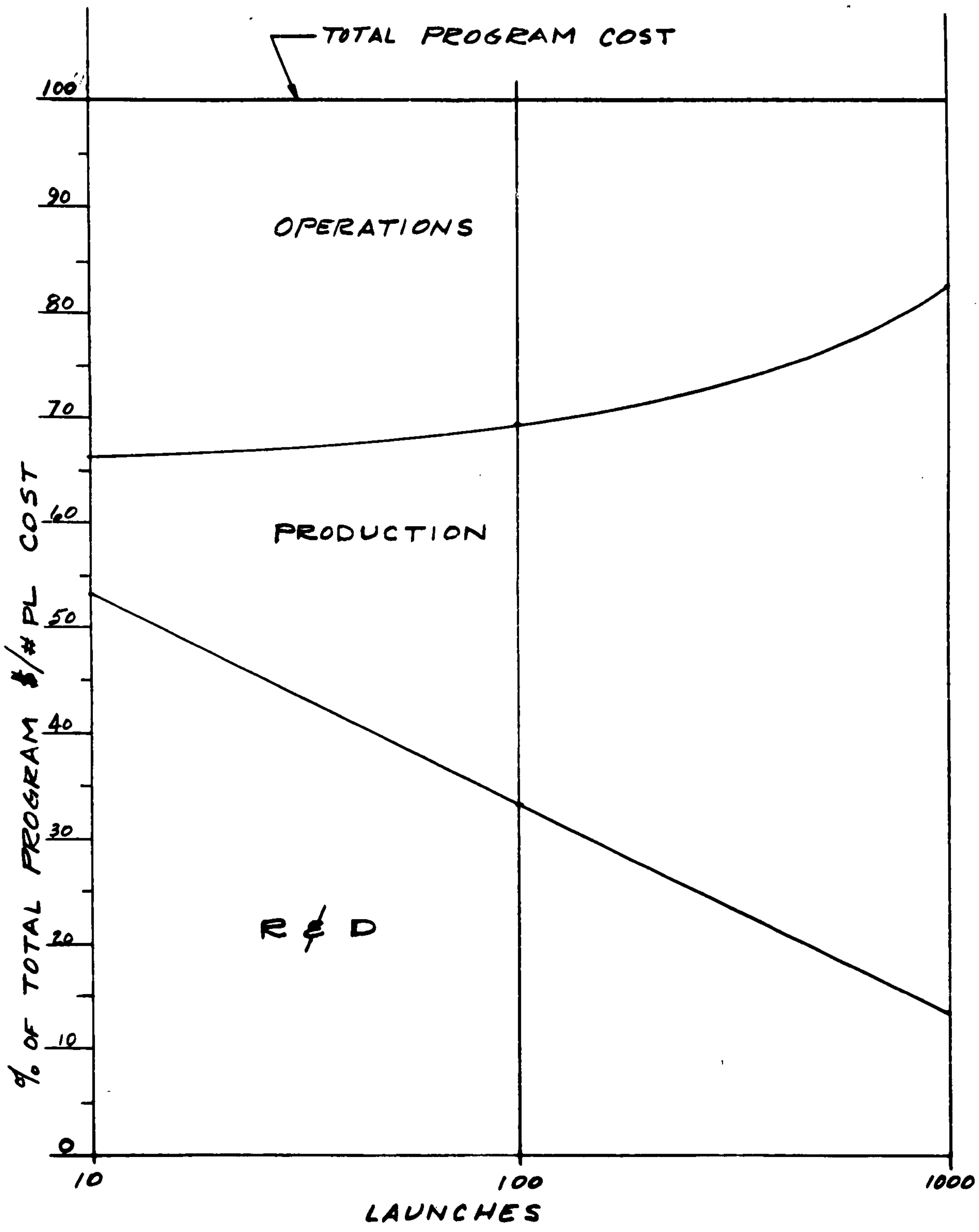


Figure 87. Total Program Cost Breakdown for 600K Advanced Systems

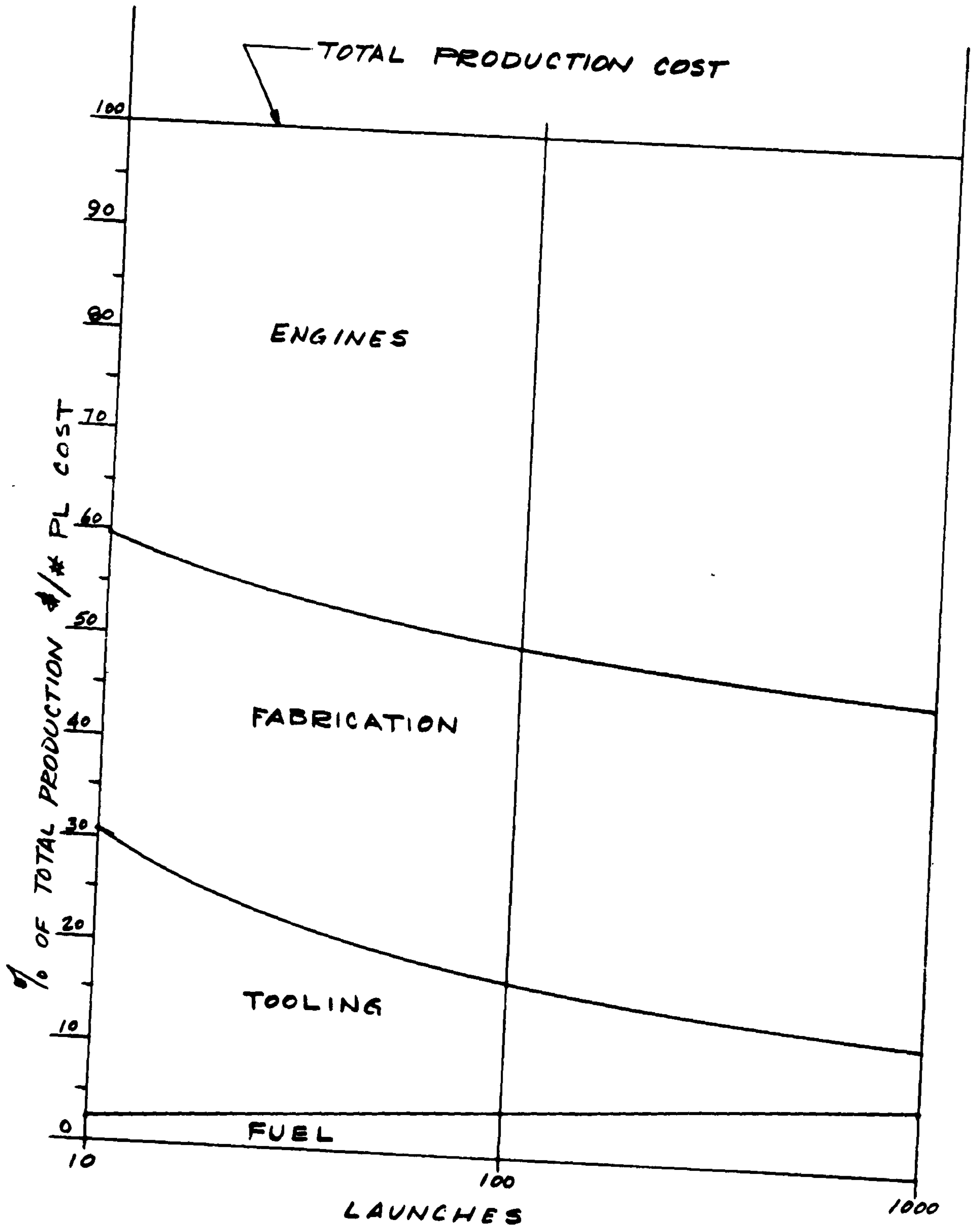


Figure 88. Total Production Cost Breakdown for 600K Advanced Systems

Table 25. Cost Optimized Booster Systems Cost Comparison

First Stage Thrust	Configuration Number	Booster System	Reliability (percent)	Payload Per Launch	Dollars Per Pound of Payload in Orbit		
					Launch Rate		
					10	100	1000
TWO-STAGE SINGLE TANK SYSTEMS							
600K	1	0.6 ER/0.2 JH	94.96	22,800	1770	270	68
600K	2	0.6 PH/0.2 JH	92.37	34,400	1290	215	60
600K	3	0.6 BH/0.2 JH	94.96	32,100	1400	220	60
600K	4	0.6 WH/0.2 JH	94.27	34,900	1330	205	57
600K	5	0.6 J'H/0.2 JH	94.98	34,600	1070	180	53
600K	6c	0.6 ER/0.3 WH	94.32	30,900	1550	230	59
600K	7c	0.6 PH/0.3 WH	91.74	40,800	1260	205	56
600K	8c	0.6 BH/0.1 WH	94.37	34,200	1360	210	58
600K	9a	0.6 WH/0.1 WH	93.63	38,700	1240	190	46
600K	9c'	2(0.3) WH/0.3 W'H	92.69	40,500	1060	175	51
800K	23	2(0.4 WH/0.4 W'H	92.20	54,300	900	150	46
800K	22	5(0.16) J''H/0.2 JH	93.20	45,300	890	155	45
1500K	10b	1.5 FR/2(0.2) JH	94.66	64,600	740	135	42
1500K	16c'	2(0.75) WH/0.75 W'H	92.69	98,000	700	130	45
1500K	11b	1.5 PR/2(0.2) JH	92.98	65,700	890	150	45
1500K	12b	1.5 WH/2(0.2) JH	93.93	94,000	780	135	45
1500K	13b	1.5 PH/2(0.2) JH	92.08	86,300	710	135	45
1500K	14b	1.5 FR/0.4 WH	94.32	73,200	760	130	40
1500K	15c	1.5 PR/0.5 WH	91.74	76,300	890	145	42
1500K	16a	1.5 WH/0.3 WH	93.63	94,000	860	145	47
1500K	17a	1.5 PH/0.3 WH	91.74	92,400	750	132	42
1500K	18c	1.5 FR/0.5 PH	92.41	67,200	850	155	48
1500K	19c	1.5 PR/0.5 PH	89.89	65,200	1030	175	52
1500K	20a	1.5 WH/0.3 PH	91.74	93,890	890	155	50
1500K	21a	1.5 PH/0.3 PH	89.89	81,800	870	160	52
3000K	24	2(0.5) FR/4(0.2) JH	93.80	138,000	510	100	34
3000K	25	4(0.75) WH/2(0.75) W'H	92.70	199,000	470	100	39

Configuration Type	Configuration Number	Booster System	Reliability (percent)	Payload Per Launch	Dollars Per Pound of Payload in Orbit		
					Launch Rate		
					10	100	1000
ALTERNATE CONCEPT SYSTEMS							
Lateral	1A	1.5 PH—DAC Lateral	80.00	70,000	560	140	48
Lateral	2A	3 3(0.167) J'''H—NAA Lateral	95.10	77,000	580	120	45
Miscellaneous	3A	2(0.3) WH/0.2 YF	93.60	41,000	1240	190	52
Miscellaneous	4A	1.4 Solid/0.3 W'H	94.40	43,000	1280	195	47
Miscellaneous	8A	0.6 W'''H (Finned)	96.02	30,700	1253	205	55
Miscellaneous	8B	0.6 W'''H (Jet)	96.02	32,800	1163	185	48
First stage cluster	6A	3 2(0.3) WH / 2(0.3) W'H	84.39	105,000	490	115	48
First stage cluster	$6A_1$	3 2(0.3) WH /2(0.3) W'H	85.39	118,000	460	113	48
First stage cluster	$6A_2$	3 2(0.3) WH /3(0.3) W'H	84.54	120,000	560	125	52
First stage cluster	7A	4 2(0.3) WH / 2(0.3) W'H	82.02	139,000	420	105	44
First stage cluster	$7A_1$	4 2(0.3) WH /3(0.3) W'H	81.20	169,000	445	105	46
First stage cluster	$7A_2$	4 2(0.3) WH /4(0.3) W'H	80.39	174,000	445	110	47
Recovery	5A	2(0.3) WH/0.3 W'H	96.04	39,500	1200	175	44
Recovery	9A	2(0.3) WH/0.05 W'H	96.04	35,200	1320	185	43
Recovery	9B	2(0.3) WH/0.1 W'H	96.04	37,400	1300	185	43
Recovery	10A	2(0.75) WH/0.75 W'H	96.04	88,300	800	120	34

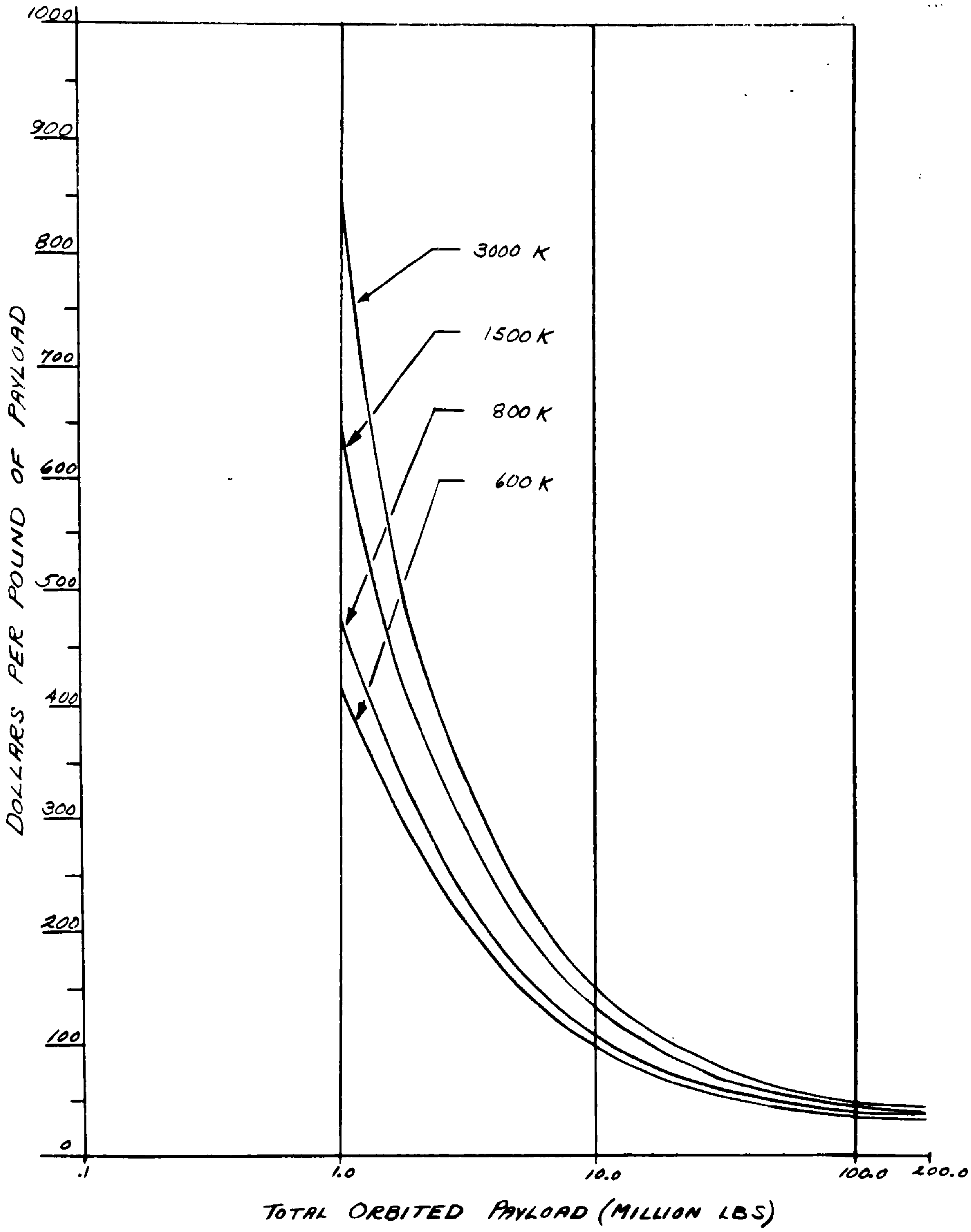


Figure 89. Dollars Per Pound of Payload Versus Total Orbited Payload

ITEMIZED COSTS OF A TYPICAL SYSTEM

Figure 90 shows the annual expenditures required for the selected typical two-stage program. The costs are spread by fiscal year for the 3 years of the R & D program and the 10 years of the operational program. These expenditures were not corrected to show the effects of the 3-1/2 percent annual inflation factor. The very high expenditure of funds in the early years of this program reflect the emphasis upon reliability growth through extensive laboratory testing. The reliability program would be coordinated with the preliminary and final design phases of engineering effort along with extensive component and system testing. This is necessary for attainment of the planned high reliability goals.

All development and production cost results presented are based upon a calculated launch reliability of less than 100 percent. The overall booster costs-per pound of payload in orbit were determined on an estimated mission reliability for each configuration. The effect of reliability upon total program cost versus number of launches for a typical two-stage booster system is shown in Figure 91.

The range of reliabilities for boosters covered in this study run from 80 through 96 percent. The effect of decreased reliability is to reduce the total payload successfully launched at any given launch rate and to raise the total cost per pound of payload. These effects are the same for all configurations presented in this study.

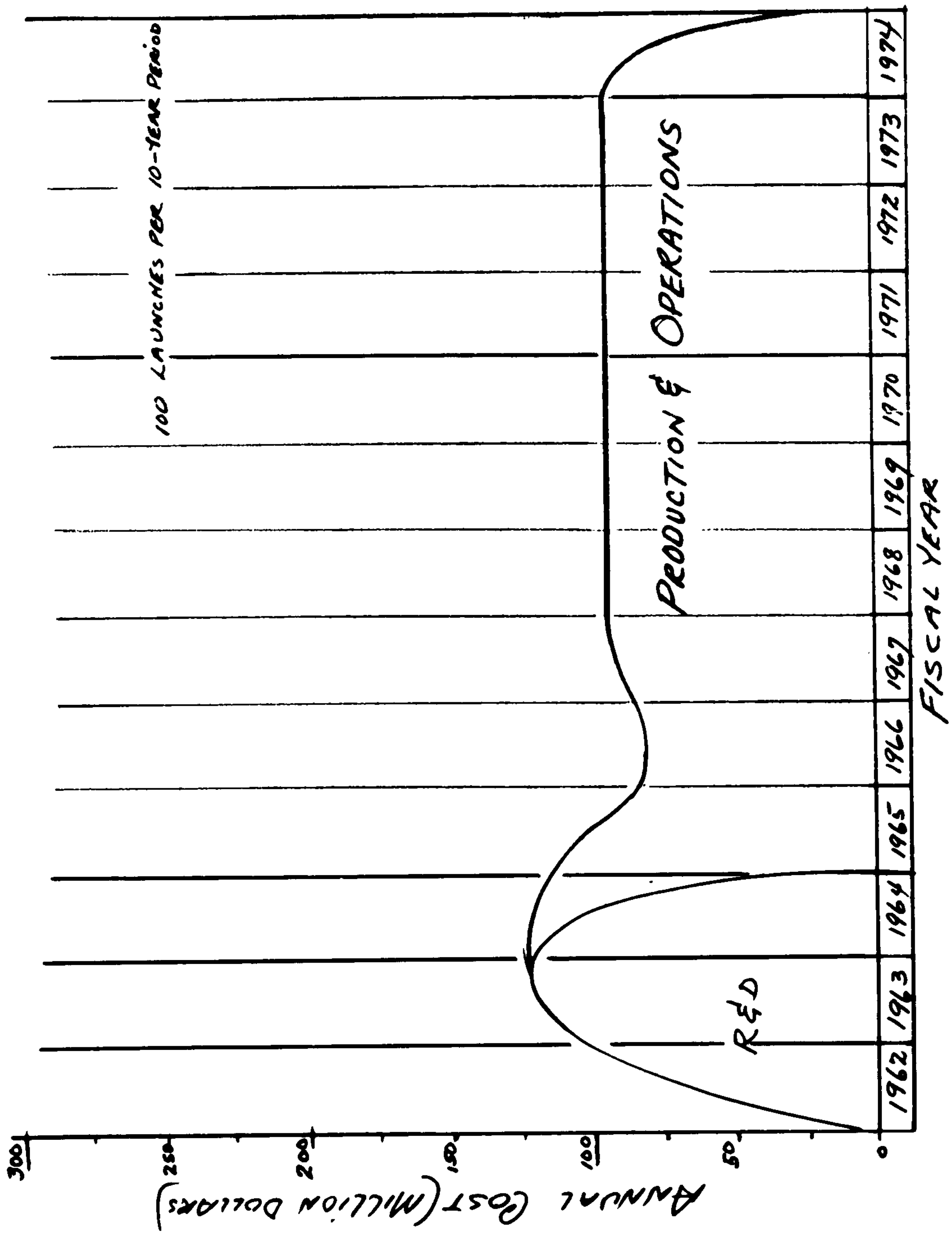


Figure 90. Annual Expenditures for Expendable 1500K Two-Stage System

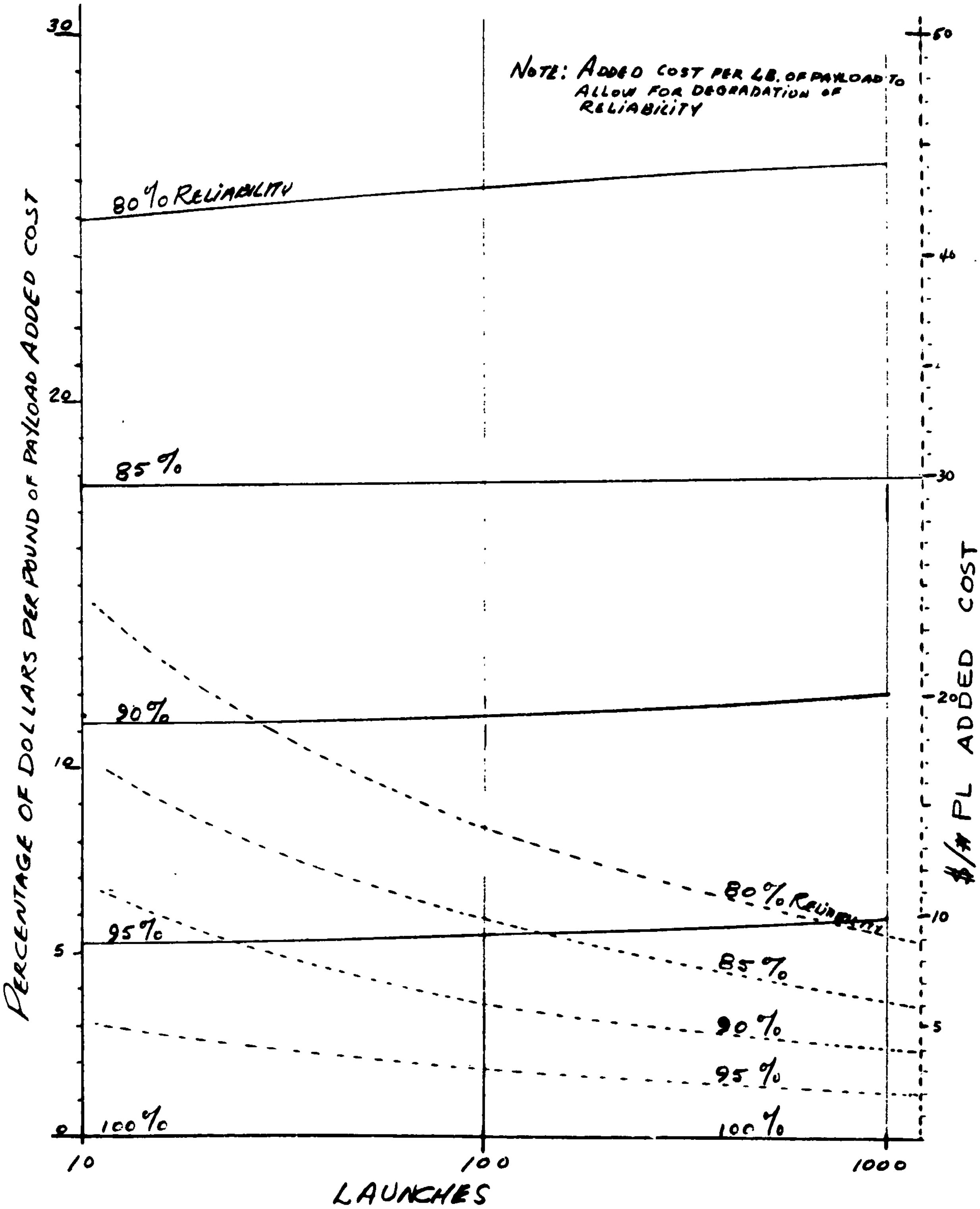


Figure 91. Effect of Reliability on 600K Two-Stage Booster System Cost

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ALTERNATE SYSTEM CONCEPTS EVALUATION

The second phase of the booster study covered comparisons of alternate configurations with the selected two-stage systems. Among the alternate configurations considered were clustered stages, lateral staging, segmented solid first stages, liquid florine second stage, paraglider first-stage recovery, single-stage-to-orbit configurations, and extended first-stage configurations.

CONFIGURATION COMPARISONS AND TRENDS

The concept of clustering modular tanks in the first stage to obtain boosters with higher thrust levels shows the best cost-per-pound-of-payload advantage at lower launch rates and at the same time has the advantage of greater program flexibility. The greatest economic advantage is noted if an advanced high-pressure propulsion system is employed. Figure 92 shows the comparative costs of the various clustered configurations. The selected two-stage 1.5 million-pound thrust vehicle is shown on the same curve as the reference vehicle or standard. The clustering cost advantage is obtained primarily from th· effect of higher production at a lower launch rate.

The cost differences of the clustered vehicles over the reference vehicle are slightly more pronounced with the common-module configurations than with the cost optimized second-stage configurations. These two-stage clustered vehicles utilize a common module in both stages. Use of advanced high-pressure engines allow efficient expansion ratios for both stages, as opposed to the normal impulse degradation in conventional bell-propulsion systems.

The laterally staged booster configurations are limited in their payload capability but are economically competitive with conventional two-stage booster systems. This advantage is most significant at the lower launch rates of 50 per 10-year period and is not reflected at the higher total launches. Figure 93 shows the comparison of the laterally staged configurations with the reference two-stage 1.5 million-pound thrust selected configuration. The reason for the economic advantage of the lateral staged configurations is that they are designed as semimodular and employ either identical tanks or engines.

Also shown on Figure 93 is the referenced two-stage 1.5 million-pound thrust configuration with its dollars-per-pound-of-payload recalculated on the basis of a paraglider recovery system for the first stage. The recoverable

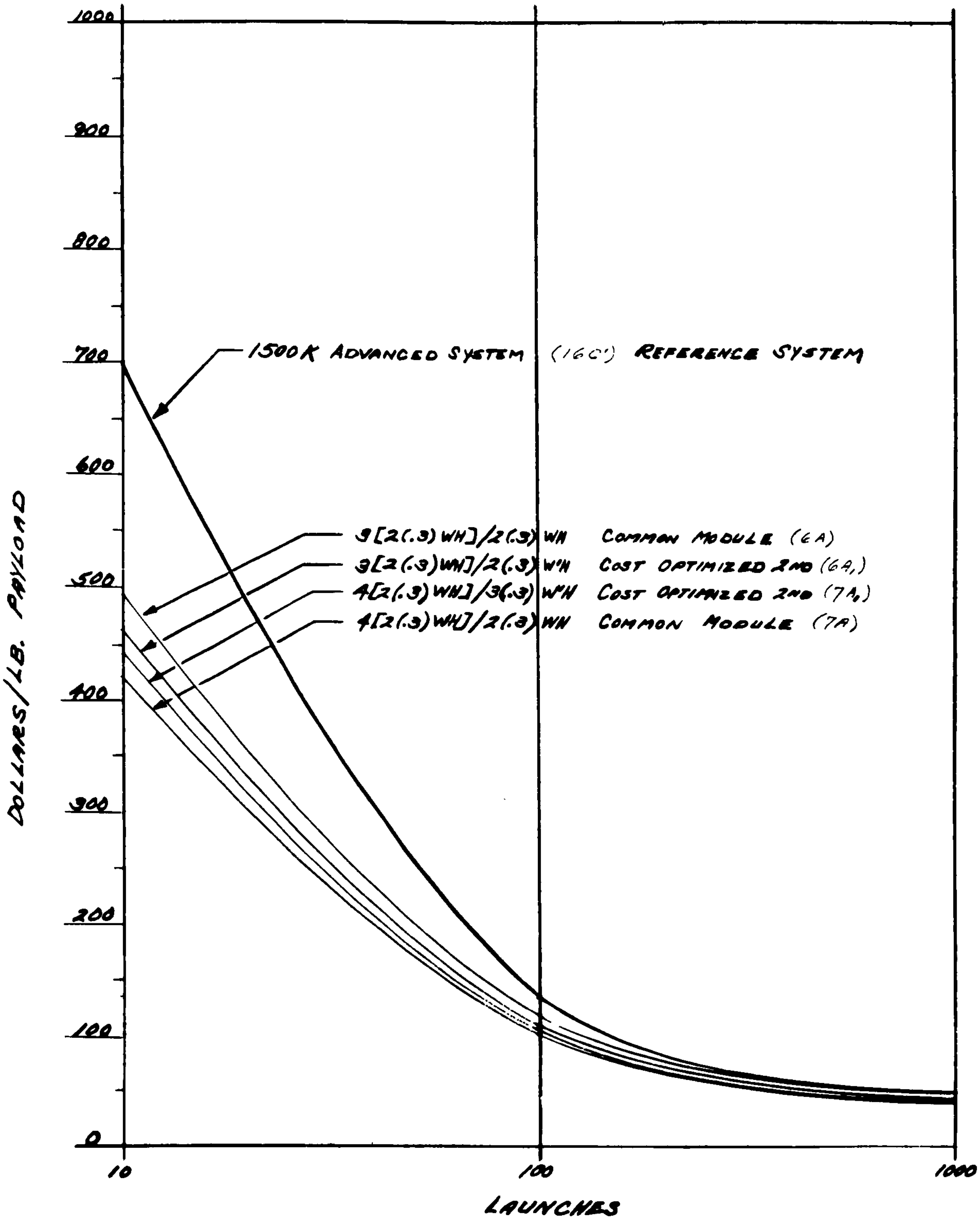


Figure 92. 600K Clustered Systems Cost Comparison - Common Module Versus Cost Optimized Stages

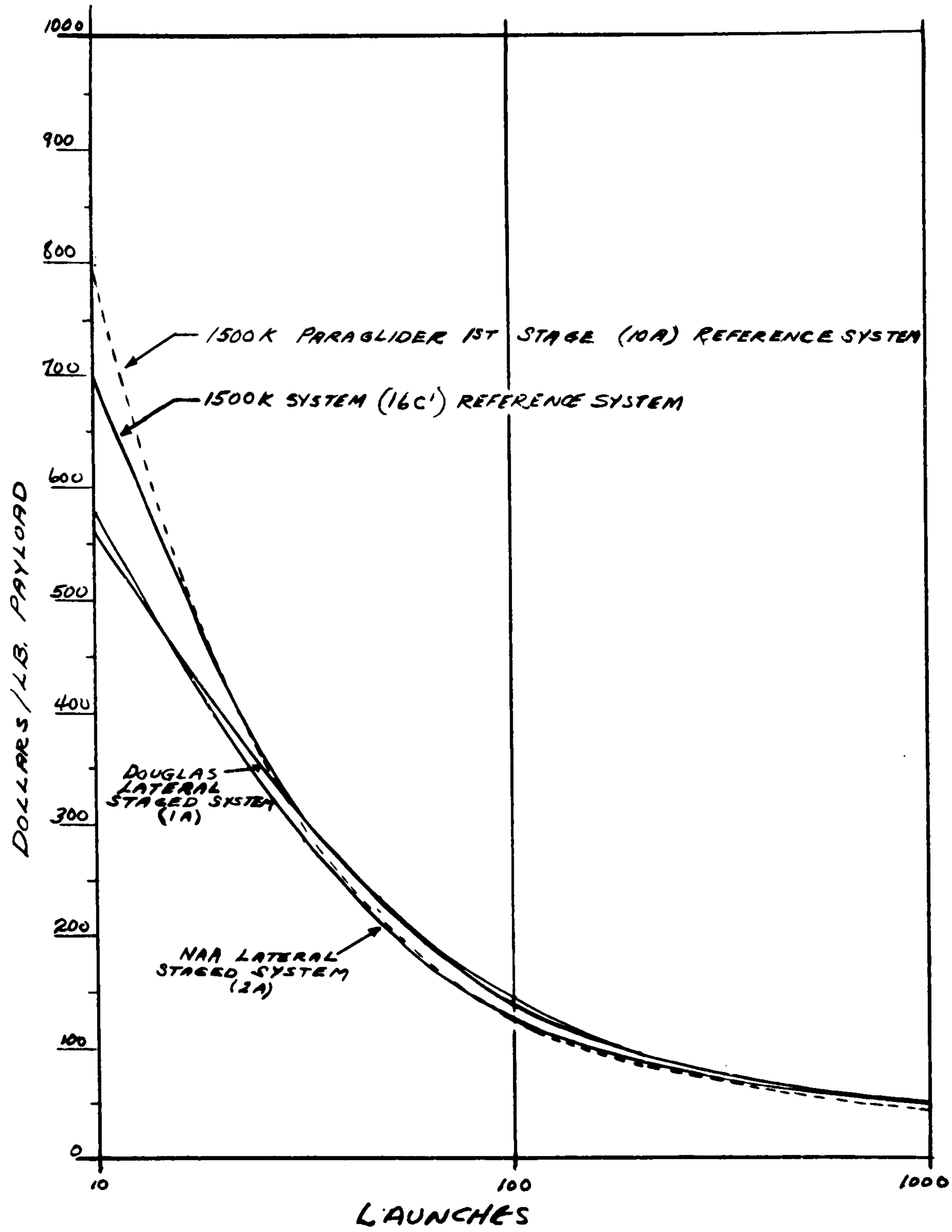


Figure 93. 1500K Lateral Staging Cost Comparison

two-stage configuration starts to show a slight cost advantage in dollars per pound of payload at the 50 per 10-year period launch rate when compared with the expendable 1.5 million-pound thrust two-stage configuration. When the recoverable two-stage configuration is compared with the laterally staged configurations, the breakeven point occurs at the 100 per 10-year period launch rate. The reason for the breakdown point occurring at a higher launch is because the expendable laterally staged vehicle has a cheaper dollars per pound of payload than the conventional two-stage system.

The dollars-per-pound-of-payload curves of the two-stage 0.6 million pound thrust vehicles shown in Figure 94 indicate the expendable and recoverable systems at this low thrust level are approximately equal in cost. The R & D and production costs of the paraglider recovery system tend to offset the savings in vehicle production costs from re-use of the same booster vehicle. Also, there is a payload penalty associated with the recovery system. At the higher thrust levels, there is a definite program cost savings when the first stage is recovered.

The extended first-stage vehicle system which utilizes paraglider recovery is also compared with the expendable 0.6 million-pound thrust vehicle. Figure 94 shows that extension of the first stage (higher first-stage increment velocity) with paraglider recovery is actually higher in cost over the conventional stage recovery concept. This results from the payload capability degrading faster than incremental first-stage savings.

One of the alternate two-stage configurations studied was a LF_2/LH_2 second-stage propulsion system on a cost optimized LO_2/LH_2 first-stage system. This configuration showed no economic or payload advantages over the selected systems. See Configuration 3a, Table 25.

The study of a segmented solid first stage coupled with the cost optimized LO_2/LH_2 second stage showed no economic advantage for boosting comparable payloads into orbit. The solid propellant propulsion costs were determined from a composite of several vendor costs, with a low, optimistic cost used where possible. All of the configurations covered in the study were considerably more economical than that utilizing a solid propellant first stage. See Configuration 4a, Table 25.

A review of the single-stage-to-orbit configurations indicates that their dollars per pound of payload are higher than the other 0.6 million-pound thrust vehicles. This is primarily due to a low payload capability and the requirement for a secondary expansion nozzle for the advanced high-pressure engines used. See Configuration 8a and 8b, Table 25.

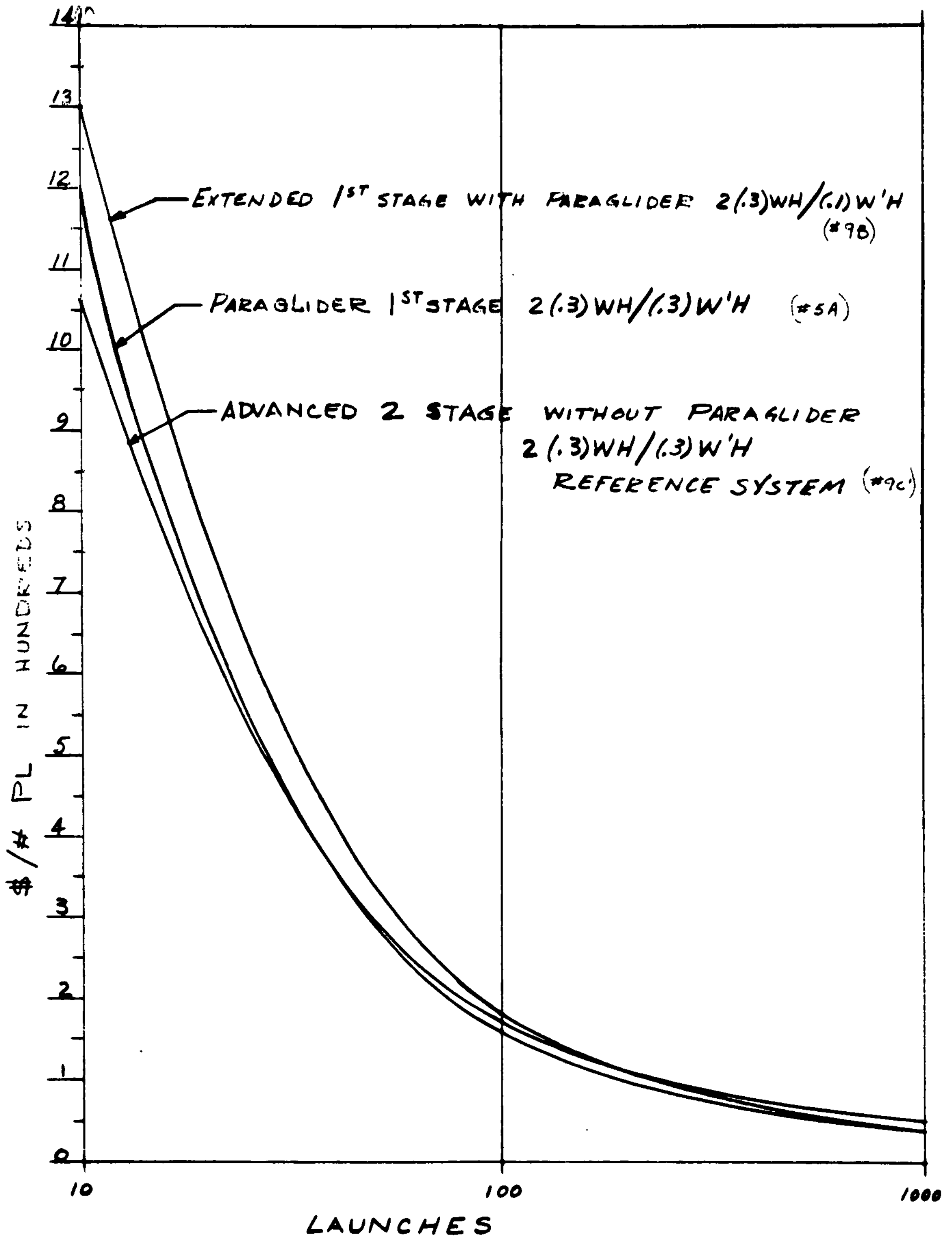


Figure 94. 600K Advanced Systems Paraglider Recovery Versus Expendable

CONCLUSIONS

In evaluating selected two-stage booster systems, it is noted that the cost per pound of payload decreases with increasing total thrust level for any given engine and propellant combination, with the greatest relative decrease occurring at the lower thrust levels of 0.6 and 0.8 million pounds of thrust. In comparing the state-of-the-art systems with the advanced high-pressure systems, it can be noted that there is no significant difference between the two within each of the thrust classes and launch rates covered. The systems utilizing existing state-of-the-art systems remain competitive with systems utilizing advanced high-pressure engines due to the reduction in R & D costs and higher initial reliability.

Tabulation of the costs for each configuration indicates that the higher the first-stage thrust level, the cheaper the dollars per pound of payload for any fixed launch rate. However, if the dollars per pound of payload orbited is compared on the basis of total pounds successfully orbited rather than launch rate, the reverse is true. This is the result of the smaller first-stage system orbiting payload earlier at a higher launch rate than the larger system, coupled with reduced prices through high production.

Two-stage boosters utilizing clustered modules in the first stage and a single tank in the second stage show a definite cost-per-pound-of-payload advantage at low launch rates. The greatest economic advantage is noted if an advanced high-pressure propulsion system is employed.

The cost differences of the clustered vehicles over single-tank vehicles are slightly more pronounced if the two-stage clustered vehicles utilize a common module in both stages. The common-module clustered booster provides one of the least expensive methods of oribiting payloads below 100 launches over a 10-year period. Generally the modular tank vehicle program costs are less than the single-tank vehicle costs at low launch rates. In very high quantity launch rates at high thrust levels, the single-tank configurations tend to become the most economical in cost per pound of payload.

The laterally staged booster configurations are limited in their payload capability, but are economically competitive with conventional two-stage booster systems because they employ either identical tanks or engines. This advantage occurs at the lower launch level of 50 per 10-year period and is not reflected at the higher total launches.

Paraglider recovery of the first stage of a conventionally staged booster represents the most economic method of orbiting payloads for launches of 100 and above over a 10-year period; a slight cost savings is realized below 100 launches in 10 years but it is not large enough to make recovery desirable.

The two-stage configuration which utilizes a LF_2/LH_2 second-stage propulsion system on a cost optimized LO_2/LH_2 first-stage system showed no economic or payload advantage over the selected system.

The study of a segmented solid coupled with the cost optimized LO_2/LH_2 second stage showed no economic advantage for boosting comparable payloads into orbit.

The dollars per pound of payload for single-stage-to-orbit configurations are higher than the selected two-stage systems. This is primarily due to a lower payload capability.

APPENDIX

The Appendix to this report contains performance, cost, and weight machine program results of selected systems configurations and are as follows: See Table 1 in Section IV for configuration code.

The Cost Optimized Booster Systems Study Engineering drawings are also included in the Appendix and are as follows:

Figure

Table A-1. .6 J'H/.2 JH Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1			
SIZING			
	TOTAL THRUST		600000.
	GROSS WEIGHT		500000.
	FUEL VOLUME		12821.
	OXIDIZER VOLUME		3974.
TANKS			
	FUEL TANK		
	INSULATION	525.	
	STRUCTURE	3350.	
	BOND AND MISC.	350.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	1062.	
	BOND AND MISC.	88.	
	TOTAL WEIGHT		5375.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	1985.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		1985.
AFT SKIRT			
	INSULATION	265.	
	STRUCTURE	1285.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		1550.
FORWARD SKIRT			
	INSULATION	280.	
	STRUCTURE	2020.	
	BOND AND MISC.	330.	
	TOTAL WEIGHT		2630.

Table A-1. .6 J'H/.2 JH Performance, Cost and Weight Machine Program Results (Cont)

MISC. TUNNELS, ETC			
TOTAL WEIGHT		240.	
RECOVERY PROVISIONS			
SURFACES	0.		
LANDING GEAR	0.		
INSULATION	0.		
TOTAL WEIGHT		0.	
PROPULSION SYSTEM			
FUEL PRESSURIZATION SYSTEM	450.		
FILL AND VENT SYSTEM	230.		
FEED SYSTEM	360.		
TOTAL ENGINE WEIGHT	8692.		
TOTAL WEIGHT		9732.	
RECOVERY SYSTEM			
LIFTING SURFACES	0.		
CONTROL SURFACES	0.		
LANDING GEAR	0.		
EXTERNAL STOWAGE COMPARTMENT	0.		
TOTAL WEIGHT		0.	
EQUIPMENT			
EQUIPMENT WEIGHT		870.	
WEIGHT EMPTY TOTAL • • •			22382.
RESIDUAL			
PRESSURANT	550.		
PROPELLANT TRAPPED IN ENGINES	508.		
PROPELLANT TRAPPED IN LINES	0.		
PROPELLANT TRAPPED IN TANK	880.		
TOTAL WEIGHT		1938.	
END BOOST TOTAL • • •			24320.
PROPELLANT			
FUEL	56667.		
OXIDIZER	283333.		
TOTAL WEIGHT		340000.	
GROSS WEIGHT / STAGE • • •			364320.

Table A-1. .6 J'H/.2 JH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	8692.		
VALVE WEIGHT	400.		
FLIGHT CONTROL WEIGHT	500.		
TOTAL WEIGHT		9592.	
AMPR			12790.
AMPR/MODULE			12790.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			340000.

Table A-1. .6 J'H/.2 JH Performance, Cost and Weight Machine Program Results (Cont)

DETAILED WEIGHT STATEMENT

STAGE 2			
SIZING			
	TOTAL THRUST		200000.
	GROSS WEIGHT		135680.
	FUEL VOLUME		3530.
	OXIDIZER VOLUME		1094.
TANKS			
	FUEL TANK		
	INSULATION	311.	
	STRUCTURE	1146.	
	BOND AND MISC.	213.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	571.	
	BOND AND MISC.	39.	
	TOTAL WEIGHT		2280.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	500.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		500.
AFT SKIRT			
	INSULATION	109.	
	STRUCTURE	452.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		561.
FORWARD SKIRT			
	INSULATION	0.	
	STRUCTURE	0.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		0.

Table A-1. .6 J'H/.2 JH Performance, Cost and Weight Machine Program Results (Cont)

MISC, TUNNELS, ETC				
	TOTAL WEIGHT		114.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	295.		
	FILL AND VENT SYSTEM	98.		
	FEED SYSTEM	100.		
	TOTAL ENGINE WEIGHT	2500.		
	TOTAL WEIGHT		2993.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		545.	
WEIGHT EMPTY TOTAL • • •				6993.
RESIDUAL				
	PRESSURANT	152.		
	PROPELLANT TRAPPED IN ENGINES	127.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	226.		
	TOTAL WEIGHT		506.	
END BOOST TOTAL • • •				7499.
PROPELLANT				
	FUEL	15603.		
	OXIDIZER	78016.		
	TOTAL WEIGHT		93619.	
GROSS WEIGHT / STAGE • • •				101118.

Table A-1. .6 J'H/.2 JH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	2500.		
VALVE WEIGHT	180.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		2980.	
AMPR			4013.
AMPR/MODULE			4013.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			93619.

TOTALS PER VEHICLE

PAYLOAD 34562.

NUMBER OF VEHICLES	COST	COST-REC FACTOR	COST/LB.PL FACTOR	COST/LB.PL-REC FACTOR
10.	3.864646E 07	0.	112.	0.
100.	2.061215E 08	0.	60.	0.
1000.	1.259229E 09	0.	36.	0.

[redacted]

Table A-2. (2).3 WH/.3 WH Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1

SIZING			
	TOTAL THRUST		600000.
	GROSS WEIGHT		500000.
	FUEL VOLUME		12443.
	OXIDIZER VOLUME		3857.
TANKS			
	FUEL TANK		
	INSULATION	515.	
	STRUCTURE	3251.	
	BOND AND MISC.	343.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	1031.	
	BOND AND MISC.	86.	
	TOTAL WEIGHT		5226.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	1985.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		1985.
AFT SKIRT			
	INSULATION	260.	
	STRUCTURE	1247.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		1507.
FORWARD SKIRT			
	INSULATION	280.	
	STRUCTURE	2020.	
	BOND AND MISC.	330.	
	TOTAL WEIGHT		2630.

Table A-2. (2).3 WH/.3 WH Performance, Cost and Weight Machine Program Results (Cont)

MISC, TUNNELS, ETC				
	TOTAL WEIGHT		235.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	437.		
	FILL AND VENT SYSTEM	223.		
	FEED SYSTEM	360.		
	TOTAL ENGINE WEIGHT	5500.		
	TOTAL WEIGHT		6520.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		857.	
WEIGHT EMPTY TOTAL * * *				18961.
RESIDUAL				
	PRESSURANT	534.		
	PROPELLANT TRAPPED IN ENGINES	550.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	854.		
	TOTAL WEIGHT		1938.	
END BOOST TOTAL * * *				20898.
PROPELLANT				
	FUEL	55000.		
	OXIDIZER	275000.		
	TOTAL WEIGHT		330000.	
GROSS WEIGHT / STAGE * * *				350898.

Table A-2. (2).3 WH/.3 WH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS CEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	5500.		
VALVE WEIGHT	300.		
FLIGHT CONTROL WEIGHT	500.		
TOTAL WEIGHT		6300.	
AMPR			12661.
AMPR/MODULE			12661.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			330000.

Table A-2. (2). 3 WH/. 3 WH Performance, Cost and Weight Machine Program Results (Cont)

DETAILED WEIGHT STATEMENT

STAGE 2

SIZING			
	TOTAL THRUST		300000.
	GROSS WEIGHT		149102.
	FUEL VOLUME		3711.
	OXIDIZER VOLUME		1150.
TANKS			
	FUEL TANK		
	INSULATION	322.	
	STRUCTURE	1204.	
	BOND AND MISC.	~~220.~~ 1746	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	601.	
	BOND AND MISC.	40.	
	TOTAL WEIGHT		2386.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	750.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		750.
AFT SKIRT			
	INSULATION	112.	
	STRUCTURE	475.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		588.
FORWARD SKIRT			
	INSULATION	0.	
	STRUCTURE	0.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		0.

Table A-2. (2). 3 WH/. 3 WH Performance, Cost and Weight Machine Program Results (Cont)

MISC, TUNNELS, ETC				
	TOTAL WEIGHT		118.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	310.		
	FILL AND VENT SYSTEM	103.		
	FEED SYSTEM	150.		
	TOTAL ENGINE WEIGHT	2500.		
	TOTAL WEIGHT		3063.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		559.	
WEIGHT EMPTY TOTAL • • •				7465.
RESIDUAL				
	PRESSURANT	160.		
	PROPELLANT TRAPPED IN ENGINES	150.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	238.		
	TOTAL WEIGHT		548.	
END BOOST TOTAL • • •				8013.
PROPELLANT				
	FUEL	16401.		
	OXIDIZER	82006.		
	TOTAL WEIGHT		98407.	
GROSS WEIGHT / STAGE • • •				106420.

Table A-2. (2). 3 WH/. 3 WH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	2500.		
VALVE WEIGHT	230.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		3030.	
AMPR			4435.
AMPR/MODULE			4435.
THRUST/WEIGHT RATIO			2.
PROPELLANT/MODULE			98407.

TOTALS PER VEHICLE

PAYLOAD 42682.

NUMBER OF VEHICLES	COST	COST-REC FACTOR	COST/LB.PL FACTOR	COST/LB.PL-REC FACTOR
10.	3.898331E 07	0.	91.	0.
100.	2.203856E 08	0.	52.	0.
1000.	1.427683E 09	0.	33.	0.

Table A-3. (5). 16 J''H/. 2 JH Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1

SIZING	TOTAL THRUST		800000.
	GROSS WEIGHT		666667.
	FUEL VOLUME		17949.
	OXIDIZER VOLUME		5563.
TANKS	FUEL TANK		
	INSULATION	657.	
	STRUCTURE	4690.	
	BOND AND MISC.	438.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	1487.	
	BOND AND MISC.	110.	
	TOTAL WEIGHT		7382.
CLUSTERING STRUCTURE	CLUSTERING STRUCTLRE WEIGHT		0.
THRUST STRUCTURE	THRUST STRUCTURE	2647.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		2647.
AFT SKIRT	INSULATION	332.	
	STRUCTURE	1799.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		2131.
FORWARD SKIRT	INSULATION	339.	
	STRUCTURE	2447.	
	BOND AND MISC.	400.	
	TOTAL WEIGHT		3186.
MISC, TUNNELS, ETC	TOTAL WEIGHT		300.

Table A-3. (5).16 J"H/.2 JH Performance, Cost and Weight Machine Program Results (Cont)

RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	630.		
	FILL AND VENT SYSTEM	322.		
	FEED SYSTEM	480.		
	TOTAL ENGINE WEIGHT	12500.		
	TOTAL WEIGHT		13932.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		1015.	
WEIGHT EMPTY TOTAL • • •				30593.
RESIDUAL				
	PRESSURANT	770.		
	PROPELLANT TRAPPED IN ENGINES	635.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	1232.		
	TOTAL WEIGHT		2637.	
END BOOST TOTAL • • •				33230.
PROPELLANT				
	FUEL	79333.		
	OXIDIZER	396667.		
	TOTAL WEIGHT		476000.	
GROSS WEIGHT / STAGE • • •				509230.
ITEMS DEDUCTED FROM EMPTY WEIGHT				
	ENGINE DRY WT. TOTAL	12500.		
	VALVE WEIGHT	450.		
	FLIGHT CONTROL WEIGHT	500.		
	TOTAL WEIGHT		13450.	
AMPR				17143.

Table A-3. (5). 16 J''H/. 2 JH Performance, Cost and Weight Machine Program Results (Cont)

AMPR/MODULE			17143.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			476000.

DETAILED WEIGHT STATEMENT

STAGE 2

SIZING			
	TOTAL THRUST		200000.
	GROSS WEIGHT		157437.
	FUEL VOLUME		3930.
	OXIDIZER VOLUME		1218.
TANKS			
	FUEL TANK		
	INSULATION	335.	
	STRUCTURE	1275.	
	BOND AND MISC.	228.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	636.	
	BOND AND MISC.	41.	
	TOTAL WEIGHT		2516.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	500.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		500.
AFT SKIRT			

Table A-3. (5).16 J"H/.2 JH Performance, Cost and Weight Machine Program Results (Cont)

	INSULATION	117.		
	STRUCTURE	504.		
	BOND AND MISC.	0.		
	TOTAL WEIGHT		620.	
FORWARD SKIRT				
	INSULATION	0.		
	STRUCTURE	0.		
	BOND AND MISC.	0.		
	TOTAL WEIGHT		0.	
MISC, TUNNELS, ETC				
	TOTAL WEIGHT		122.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	328.		
	FILL AND VENT SYSTEM	109.		
	FEED SYSTEM	100.		
	TOTAL ENGINE WEIGHT	2500.		
	TOTAL WEIGHT		3038.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		575.	
WEIGHT EMPTY TOTAL • • •				7371.
RESIDUAL				
	PRESSURANT	170.		
	PROPELLANT TRAPPED IN ENGINES	127.		
	PROPELLANT TRAPPED IN LINES	0.		

Table A-3. (5). 16 J''H/. 2 JH Performance, Cost and Weight Machine Program Results (Cont)

PROPELLANT TRAPPED IN TANK	252.		
TOTAL WEIGHT		548.	
END BOOST TOTAL • • •			7920.
PROPELLANT			
FUEL	17371.		
OXIDIZER	86853.		
TOTAL WEIGHT		104223.	
GROSS WEIGHT / STAGE • • •			112143.
ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	2500.		
VALVE WEIGHT	180.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		2980.	
AMPR			4391.
AMPR/MODULE			4391.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			104223.

TOTALS PER VEHICLE

PAYLOAD 45294.

NUMBER OF VEHICLES	COST	COST-REC FACTOR	COST/LB.PL FACTOR	COST/LB.PL-REC FACTOR
10.	4.755510E 07	0.	105.	0.
100.	2.519125E 08	0.	56.	0.
1000.	1.534392E 09	0.	34.	0.

Table A-4. (2).4 WH/.4 W'H Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1

SIZING	TOTAL THRUST		800000.
	GROSS WEIGHT		666667.
	FUEL VOLUME		17647.
	OXIDIZER VOLUME		5470.
TANKS	FUEL TANK		
	INSULATION	650.	
	STRUCTURE	4611.	
	BOND AND MISC.	433.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	1462.	
	BOND AND MISC.	109.	
	TOTAL WEIGHT		7265.
CLUSTERING STRUCTURE	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE	THRUST STRUCTURE	2647.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		2647.
AFT SKIRT	INSULATION	328.	
	STRUCTURE	1769.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		2097.
FORWARD SKIRT	INSULATION	339.	
	STRUCTURE	2447.	
	BOND AND MISC.	400.	
	TOTAL WEIGHT		3186.

Table A-4. (2).4 WH/.4 W'H Performance, Cost and Weight Machine Program Results (Cont)

MISC, TUNNELS, ETC				
TOTAL WEIGHT			297.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
TOTAL WEIGHT			0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	619.		
	FILL AND VENT SYSTEM	317.		
	FEED SYSTEM	480.		
	TOTAL ENGINE WEIGHT	6000.		
TOTAL WEIGHT			7416.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
TOTAL WEIGHT			0.	
EQUIPMENT				
EQUIPMENT WEIGHT			1008.	
WEIGHT EMPTY TOTAL • • •				23915.
RESIDUAL				
	PRESSURANT	757.		
	PROPELLANT TRAPPED IN ENGINES	600.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	1211.		
TOTAL WEIGHT			2568.	
END BOOST TOTAL • • •				26483.
PROPELLANT				
	FUEL	78000.		
	OXIDIZER	390000.		
TOTAL WEIGHT			468000.	
GROSS WEIGHT / STAGE • • •				494483.

Table A-4. (2).4 WH/.4 W'H Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	6000.		
VALVE WEIGHT	500.		
FLIGHT CONTROL WEIGHT	500.		
TOTAL WEIGHT		7000.	
AMPR			16915.
AMPR/MODULE			16915.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			468000.

Table A-4. (2).4 WH/.4 W'H Performance, Cost and Weight Machine Program Results (Cont)

DETAILED WEIGHT STATEMENT

STAGE 2

Group	Item	Component	Subtotal
SIZING	TOTAL THRUST		400000.
	GROSS WEIGHT		172184.
	FUEL VOLUME		4090.
	OXIDIZER VOLUME		1268.
TANKS	FUEL TANK		
	INSULATION	344.	
	STRUCTURE	1327.	
	BOND AND MISC.	235.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	662.	
	BOND AND MISC.	43.	
	TOTAL WEIGHT		2610.
CLUSTERING STRUCTURE	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE	THRUST STRUCTURE	1000.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		1000.
AFT SKIRT	INSULATION	120.	
	STRUCTURE	524.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		644.
FORWARD SKIRT	INSULATION	0.	
	STRUCTURE	0.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		0.

Table A-4. (2).4 WH/.4 W'H Performance, Cost and Weight Machine Program Results (Cont)

MISC. TUNNELS, ETC				
	TOTAL WEIGHT		125.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	342.		
	FILL AND VENT SYSTEM	114.		
	FEED SYSTEM	200.		
	TOTAL ENGINE WEIGHT	3000.		
	TOTAL WEIGHT		3656.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		586.	
WEIGHT EMPTY TOTAL . . .				8621.
RESIDUAL				
	PRESSURANT	177.		
	PROPELLANT TRAPPED IN ENGINES	300.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	262.		
	TOTAL WEIGHT		739.	
END BOOST TOTAL . . .				9360.
PROPELLANT				
	FUEL	18079.		
	OXIDIZER	90396.		
	TOTAL WEIGHT		108476.	
GROSS WEIGHT / STAGE . . .				117836.

Table A-4. (2).4 WH/.4 W'H Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	3000.		
VALVE WEIGHT	280.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		3580.	
AMPR			5041.
AMPR/MODULE			5041.
THRUST/WEIGHT RATIO			2.
PROPELLANT/MODULE			108476.

TOTALS PER VEHICLE

PAYLOAD 54348.

NUMBER OF VEHICLES	COST	COST-REC FACTOR	COST/LB.PL FACTOR	COST/LB.PL-REC FACTOR
10.	4.841014E 07	0.	89.	0.
100.	2.677612E 08	0.	49.	0.
1000.	1.731964E 09	0.	32.	0.

Table A-5. 1.5 FR/(2). 2 JH Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1

SIZING			
	TOTAL THRUST		1500000.
	GROSS WEIGHT		1250000.
	FUEL VOLUME		5632.
	OXIDIZER VOLUME		8957.
TANKS			
	FUEL TANK		
	INSULATION	417.	
	STRUCTURE	7467.	
	BOND AND MISC.	128.	
	OXIDIZER TANK		
	INSULATION	407.	
	STRUCTURE	6418.	
	BOND AND MISC.	113.	
	TOTAL WEIGHT		14950.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	1650.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		1650.
AFT SKIRT			
	INSULATION	309.	
	STRUCTURE	836.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		1145.
FORWARD SKIRT			
	INSULATION	250.	
	STRUCTURE	1970.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		2220.

Table A-5. 1.5 FR/(2).2 JH Performance, Cost and Weight Machine Program Results (Cont)

MISC. TUNNELS, ETC				
	TOTAL WEIGHT		0.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	612.		
	FILL AND VENT SYSTEM	58.		
	FEED SYSTEM	1705.		
	TOTAL ENGINE WEIGHT	11590.		
	TOTAL WEIGHT		13965.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		296.	
WEIGHT EMPTY TOTAL • • •				34226.
RESIDUAL				
	PRESSURANT	3080.		
	PROPELLANT TRAPPED IN ENGINES	3150.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	3469.		
	TOTAL WEIGHT		9699.	
END BOOST TOTAL • • •				43925.
PROPELLANT				
	FUEL	283846.		
	OXIDIZER	638654.		
	TOTAL WEIGHT		922500.	
GROSS WEIGHT / STAGE • • •				966425.

Table A-5. 1.5 FR/(2). 2 JH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	11590.		
VALVE WEIGHT	200.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		12090.	
AMPR			22136.
AMPR/MODULE			22136.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			922500.

Table A-5. 1.5 FR/(2). 2 JH Performance, Cost and Weight Machine Program Results (Cont)

DETAILED WEIGHT STATEMENT

STAGE 2

SIZING			
	TOTAL THRUST		400000.
	GROSS WEIGHT		283575.
	FUEL VOLUME		7699.
	OXIDIZER VOLUME		2386.
TANKS			
	FUEL TANK		
	INSULATION	524.	
	STRUCTURE	2499.	
	BOND AND MISC.	358.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	1246.	
	BOND AND MISC.	65.	
	TOTAL WEIGHT		4691.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	1000.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		1000.
AFT SKIRT			
	INSULATION	183.	
	STRUCTURE	987.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		1169.
FORWARD SKIRT			
	INSULATION	0.	
	STRUCTURE	0.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		0.

Table A-5. 1.5 FR/(2). 2 JH Performance, Cost and Weight Machine Program Results (Cont)

MISC. TUNNELS, ETC				
	TOTAL WEIGHT		191.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	643.		
	FILL AND VENT SYSTEM	214.		
	FEED SYSTEM	200.		
	TOTAL ENGINE WEIGHT	5000.		
	TOTAL WEIGHT		6058.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		750.	
WEIGHT EMPTY TOTAL . . .				13860.
RESIDUAL				
	PRESSURANT	332.		
	PROPELLANT TRAPPED IN ENGINES	254.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	493.		
	TOTAL WEIGHT		1080.	
END BOOST TOTAL . . .				14939.
PROPELLANT				
	FUEL	34029.		
	OXIDIZER	170145.		
	TOTAL WEIGHT		204174.	
GROSS WEIGHT / STAGE . . .				219113.

Table A-5. 1.5 FR/(2). 2 JH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT				
ENGINE DRY WT. TOTAL	5000.			
VALVE WEIGHT	250.			
FLIGHT CONTROL WEIGHT	300.			
TOTAL WEIGHT		5550.		
AMPR			8310.	
AMPR/MODULE			8310.	
THRUST/WEIGHT RATIO			1.	
PROPELLANT/MODULE			204174.	

TOTALS PER VEHICLE

PAYLOAD 64462.

NUMBER OF VEHICLES	COST	COST-REC FACTOR	COST/LB.PL FACTOR	COST/LB.PL-REC FACTOR
10.	6.301226E 07	0.	98.	0.
100.	3.258583E 08	0.	51.	0.
1000.	1.949054E 09	0.	30.	0.

Table A-6. (2).75 WH/.75 WH Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1

SIZING			
	TOTAL THRUST		1500000.
	GROSS WEIGHT		1250000.
	FUEL VOLUME		32051.
	OXIDIZER VOLUME		9935.
TANKS			
	FUEL TANK		
	INSULATION	1122.	
	STRUCTURE	19648.	
	BOND AND MISC.	0.	
	OXIDIZER TANK		
	INSULATION	613.	
	STRUCTURE	8718.	
	BOND AND MISC.	374.	
	TOTAL WEIGHT		30475.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	1855.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		1855.
AFT SKIRT			
	INSULATION	349.	
	STRUCTURE	947.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		1296.
FORWARD SKIRT			
	INSULATION	420.	
	STRUCTURE	3350.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		3770.

Table A-6. (2).75 WH/.75 WH Performance, Cost and Weight Machine Program Results (Cont)

MISC, TUNNELS, ETC				
	TOTAL WEIGHT		0.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	528.		
	FILL AND VENT SYSTEM	60.		
	FEED SYSTEM	1600.		
	TOTAL ENGINE WEIGHT	15200.		
	TOTAL WEIGHT		17388.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		799.	
WEIGHT EMPTY TOTAL • • •				55583.
RESIDUAL				
	PRESSURANT	3004.		
	PROPELLANT TRAPPED IN ENGINES	1200.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	5305.		
	TOTAL WEIGHT		9509.	
BND BOOST TOTAL • • •				65092.
PROPELLANT				
	FUEL	141667.		
	OXIDIZER	708333.		
	TOTAL WEIGHT		850000.	
GROSS WEIGHT / STAGE • • •				915092.

Table A-6. (2).75 WH/.75 WH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT				
ENGINE DRY WT. TOTAL	15200.			
VALVE WEIGHT	800.			
FLIGHT CONTROL WEIGHT	500.			
TOTAL WEIGHT		16500.		
AMPR			39083.	
AMPR/MODULE			39083.	
THRUST/WEIGHT RATIO			1.	
PROPELLANT/MODULE			850000.	

Table A-6. (2).75 WH/.75 WH Performance, Cost and Weight Machine Program Results (Cont)

DETAILED WEIGHT STATEMENT

STAGE 2

SIZING			
	TOTAL THRUST		750000.
	GROSS WEIGHT		334908.
	FUEL VOLUME		8411.
	OXIDIZER VOLUME		2607.
TANKS			
	FUEL TANK		
	INSULATION	556.	
	STRUCTURE	2729.	
	BOND AND MISC.	379.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	1361.	
	BOND AND MISC.	69.	
	TOTAL WEIGHT		5094.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	1875.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		1875.
AFT SKIRT			
	INSULATION	194.	
	STRUCTURE	1078.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		1272.
FORWARD SKIRT			
	INSULATION	0.	
	STRUCTURE	0.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		0.

Table A-6. (2).75 WH/.75 WH Performance, Cost and Weight Machine Program Results (Cont)

MISC, TUNNELS, ETC				
TOTAL WEIGHT			203.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
TOTAL WEIGHT			0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	703.		
	FILL AND VENT SYSTEM	234.		
	FEED SYSTEM	375.		
	TOTAL ENGINE WEIGHT	7600.		
TOTAL WEIGHT			8912.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
TOTAL WEIGHT			0.	
EQUIPMENT				
EQUIPMENT WEIGHT			771.	
WEIGHT EMPTY TOTAL • • •				18127.
RESIDUAL				
	PRESSURANT	363.		
	PROPELLANT TRAPPED IN ENGINES	600.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	539.		
TOTAL WEIGHT			1502.	
END BOOST TOTAL • • •				19629.
PROPELLANT				
	FUEL	37175.		
	OXIDIZER	185874.		
TOTAL WEIGHT			223049.	
GROSS WEIGHT / STAGE • • •				242678.

Table A-6. (2).75 WH/.75 WH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	7600.		
VALVE WEIGHT	400.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		8300.	
AMPR			9827.
AMPR/MODULE			9827.
THRUST/WEIGHT RATIO			2.
PROPELLANT/MODULE			223049.

TOTALS PER VEHICLE

PAYLOAD 92230.

NUMBER OF VEHICLES	COST	COST-REC FACTOR	COST/LB.PL FACTOR	COST/LB.PL-REC FACTOR
10.	9.433677E 07	0.	102.	0.
100.	5.126821E 08	0.	56.	0.
1000.	~~1.773026E~~ 09	0.	~~40.~~	0.
	3 258025 09		36	

[redacted]

Table A-7. (2)1.5 FR/(4).2 JH Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1

SIZING			
	TOTAL THRUST		3000000.
	GROSS WEIGHT		2500000.
	FUEL VOLUME		10562.
	OXIDIZER VOLUME		16798.
TANKS			
	FUEL TANK		
	INSULATION	481.	
	STRUCTURE	12177.	
	BOND AND MISC.	179.	
	OXIDIZER TANK		
	INSULATION	1914.	
	STRUCTURE	11267.	
	BOND AND MISC.	372.	
	TOTAL WEIGHT		26391.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	4370.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		4370.
AFT SKIRT			
	INSULATION	618.	
	STRUCTURE	8433.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		9051.
FORWARD SKIRT			
	INSULATION	440.	
	STRUCTURE	3480.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		3920.
MISC, TUNNELS, ETC			
	TOTAL WEIGHT		0.

Table A-7. (2)1.5 FR/(4).2 JH Performance, Cost and Weight Machine Program Results (Cont)

RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	1058.		
	FILL AND VENT SYSTEM	69.		
	FEED SYSTEM	2335.		
	TOTAL ENGINE WEIGHT	23180.		
	TOTAL WEIGHT		26641.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		956.	
WEIGHT EMPTY TOTAL • • •				71329.
RESIDUAL				
	PRESSURANT	5819.		
	PROPELLANT TRAPPED IN ENGINES	6300.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	9967.		
	TOTAL WEIGHT		22086.	
END BOOST TOTAL • • •				93415.
PROPELLANT				
	FUEL	532308.		
	OXIDIZER	1197692.		
	TOTAL WEIGHT		1730000.	
GROSS WEIGHT / STAGE • • •				1823415.
ITEMS DEDUCTED FROM EMPTY WEIGHT				
	ENGINE DRY WT. TOTAL	23180.		
	VALVE WEIGHT	500.		
	FLIGHT CONTROL WEIGHT	300.		
	TOTAL WEIGHT		23980.	
AMPR				47349.

Table A-7. (2)1.5 FR/(4).2 JH Performance, Cost and Weight Machine Program Results (Cont)

AMPR/MODULE			
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			1730000.
DETAILED WEIGHT STATEMENT			
STAGE 2			
SIZING			
	TOTAL THRUST		800000.
	GROSS WEIGHT		676585.
	FUEL VOLUME		19364.
	OXIDIZER VOLUME		6002.
TANKS			
	FUEL TANK		
	INSULATION	691.	
	STRUCTURE	5060.	
	BOND AND MISC.	461.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	1604.	
	BOND AND MISC.	116.	
	TOTAL WEIGHT		7931.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	2647.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		2647.
AFT SKIRT			

Table A-7. (2)1.5 FR/(4).2 JH Performance, Cost and Weight Machine Program Results (Cont)

INSULATION	0.		
STRUCTURE	0.		
BOND AND MISC.	0.		
TOTAL WEIGHT		0.	
FORWARD SKIRT			
INSULATION	0.		
STRUCTURE	0.		
BOND AND MISC.	0.		
TOTAL WEIGHT		0.	
MISC, TUNNELS, ETC			
TOTAL WEIGHT		316.	
RECOVERY PROVISIONS			
SURFACES	0.		
LANDING GEAR	0.		
INSULATION	0.		
TOTAL WEIGHT		0.	
PROPULSION SYSTEM			
FUEL PRESSURIZATION SYSTEM	680.		
FILL AND VENT SYSTEM	347.		
FEED SYSTEM	460.		
TOTAL ENGINE WEIGHT	8692.		
TOTAL WEIGHT		10199.	
RECOVERY SYSTEM			
LIFTING SURFACES	0.		
CONTROL SURFACES	0.		
LANDING GEAR	0.		
EXTERNAL STOWAGE COMPARTMENT	0.		
TOTAL WEIGHT		0.	
EQUIPMENT			
EQUIPMENT WEIGHT		1047.	
WEIGHT EMPTY TOTAL * * *			22140.
RESIDUAL			
PRESSURANT	831.		
PROPELLANT TRAPPED IN ENGINES	508.		
PROPELLANT TRAPPED IN LINES	0.		

[redacted]

Table A-7. (2)1.5 FR/(4).2 JH Performance, Cost and Weight Machine Program Results (Cont)

PROPELLANT TRAPPED IN TANK	1529.		
TOTAL WEIGHT		2668.	
END BOOST TOTAL • • •			24808.
PROPELLANT			
FUEL	85588.		
OXIDIZER	427940.		
TOTAL WEIGHT		513528.	
GROSS WEIGHT / STAGE • • •			538336.
ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	8692.		
VALVE WEIGHT	400.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		9392.	
AMPR			12748.
AMPR/MODULE			12748.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			513528.

Table 7

TOTALS PER VEHICLE

PAYLOAD 138249.

NUMBER OF VEHICLES	COST	COST-REC FACTOR	COST/LB.PL FACTOR	COST/LB.PL-REC FACTOR
10.	1.188632E 08	0.	86.	0.
100.	6.1091[illegible]E 08	0.	44.	0.
1000.	3.641064E 09	0.	26.	0.

[redacted]

Table A-8. (4).75 WH/(2).75 WH Performance, Cost and Weight Machine Program Results

DETAILED WEIGHT STATEMENT

STAGE 1

SIZING				
	TOTAL THRUST			3000000.
	GROSS WEIGHT			2500000.
	FUEL VOLUME			63160.
	OXIDIZER VOLUME			19577.
TANKS				
	FUEL TANK			
		INSULATION	1764.	
		STRUCTURE	38718.	
		BOND AND MISC.	0.	
	OXIDIZER TANK			
		INSULATION	964.	
		STRUCTURE	17179.	
		BOND AND MISC.	588.	
	TOTAL WEIGHT			59213.
CLUSTERING STRUCTURE				
	CLUSTERING STRUCTURE WEIGHT			0.
THRUST STRUCTURE				
		THRUST STRUCTURE	3710.	
		CONE STRUCTURE	0.	
		CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT			3710.
AFT SKIRT				
		INSULATION	549.	
		STRUCTURE	1865.	
		BOND AND MISC.	0.	
	TOTAL WEIGHT			2414.
FORWARD SKIRT				
		INSULATION	667.	
		STRUCTURE	5318.	
		BOND AND MISC.	0.	
	TOTAL WEIGHT			5985.

Table A-8. (4).75 WH/(2).75 WH Performance, Cost and Weight Machine Program Results (Cont)

MISC, TUNNELS, ETC				
	TOTAL WEIGHT		0.	
RECOVERY PROVISIONS				
	SURFACES	0.		
	LANDING GEAR	0.		
	INSULATION	0.		
	TOTAL WEIGHT		0.	
PROPULSION SYSTEM				
	FUEL PRESSURIZATION SYSTEM	1041.		
	FILL AND VENT SYSTEM	118.		
	FEED SYSTEM	3200.		
	TOTAL ENGINE WEIGHT	30000.		
	TOTAL WEIGHT		34358.	
RECOVERY SYSTEM				
	LIFTING SURFACES	0.		
	CONTROL SURFACES	0.		
	LANDING GEAR	0.		
	EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT		0.	
EQUIPMENT				
	EQUIPMENT WEIGHT		1060.	
WEIGHT EMPTY TOTAL . . .				106740.
RESIDUAL				
	PRESSURANT	5920.		
	PROPELLANT TRAPPED IN ENGINES	2400.		
	PROPELLANT TRAPPED IN LINES	0.		
	PROPELLANT TRAPPED IN TANK	10455.		
	TOTAL WEIGHT		18774.	
END BOOST TOTAL . . .				125515.
PROPELLANT				
	FUEL	279167.		
	OXIDIZER	1395833.		
	TOTAL WEIGHT		1675000.	
GROSS WEIGHT / STAGE . . .				1800515.

Table A-8. (4).75 WH/(2).75 WH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	30000.		
VALVE WEIGHT	600.		
FLIGHT CONTROL WEIGHT	500.		
TOTAL WEIGHT		31100.	
AMPR			75640.
AMPR/MODULE			75640.
THRUST/WEIGHT RATIO			1.
PROPELLANT/MODULE			1675000.

[redacted]

Table A-8. (4).75 WH/(2).75 WH Performance, Cost and Weight Machine Program Results (Cont)

DETAILED WEIGHT STATEMENT

STAGE 2

SIZING			
	TOTAL THRUST		1500000.
	GROSS WEIGHT		699485.
	FUEL VOLUME		17408.
	OXIDIZER VOLUME		5396.
TANKS			
	FUEL TANK		
	INSULATION	902.	
	STRUCTURE	5649.	
	BOND AND MISC.	616.	
	OXIDIZER TANK		
	INSULATION	0.	
	STRUCTURE	2817.	
	BOND AND MISC.	112.	
	TOTAL WEIGHT		10097.
CLUSTERING STRUCTURE			
	CLUSTERING STRUCTURE WEIGHT		0.
THRUST STRUCTURE			
	THRUST STRUCTURE	3750.	
	CONE STRUCTURE	0.	
	CONE BOND AND/OR INSULATION	0.	
	TOTAL WEIGHT		3750.
AFT SKIRT			
	INSULATION	315.	
	STRUCTURE	2231.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		2546.
FORWARD SKIRT			
	INSULATION	0.	
	STRUCTURE	0.	
	BOND AND MISC.	0.	
	TOTAL WEIGHT		0.

Table A-8. (4). 75 WH/(2). 75 WH Performance, Cost and Weight Machine Program Results (Cont)

MISC. TUNNELS, ETC					
	TOTAL WEIGHT			329.	
RECOVERY PROVISIONS					
		SURFACES	0.		
		LANDING GEAR	0.		
		INSULATION	0.		
	TOTAL WEIGHT			0.	
PROPULSION SYSTEM					
		FUEL PRESSURIZATION SYSTEM	1455.		
		FILL AND VENT SYSTEM	485.		
		FEED SYSTEM	750.		
		TOTAL ENGINE WEIGHT	15000.		
	TOTAL WEIGHT			17690.	
RECOVERY SYSTEM					
		LIFTING SURFACES	0.		
		CONTROL SURFACES	0.		
		LANDING GEAR	0.		
		EXTERNAL STOWAGE COMPARTMENT	0.		
	TOTAL WEIGHT			0.	
EQUIPMENT					
	EQUIPMENT WEIGHT			912.	
WEIGHT EMPTY TOTAL • • •					35324.
RESIDUAL					
		PRESSURANT	752.		
		PROPELLANT TRAPPED IN ENGINES	1200.		
		PROPELLANT TRAPPED IN LINES	0.		
		PROPELLANT TRAPPED IN TANK	1115.		
	TOTAL WEIGHT			3067.	
END BOOST TOTAL • • •					38391.
PROPELLANT					
		FUEL	76943.		
		OXIDIZER	384717.		
	TOTAL WEIGHT			461660.	
GROSS WEIGHT / STAGE • • •					500051.

[redacted]

Table A-8. (4).75 WH/(2).75 WH Performance, Cost and Weight Machine Program Results (Cont)

ITEMS DEDUCTED FROM EMPTY WEIGHT			
ENGINE DRY WT. TOTAL	15000.		
VALVE WEIGHT	300.		
FLIGHT CONTROL WEIGHT	300.		
TOTAL WEIGHT		15600.	
AMPR			19724.
AMPR/MODULE			19724.
THRUST/WEIGHT RATIO			2.
PROPELLANT/MODULE			461660.

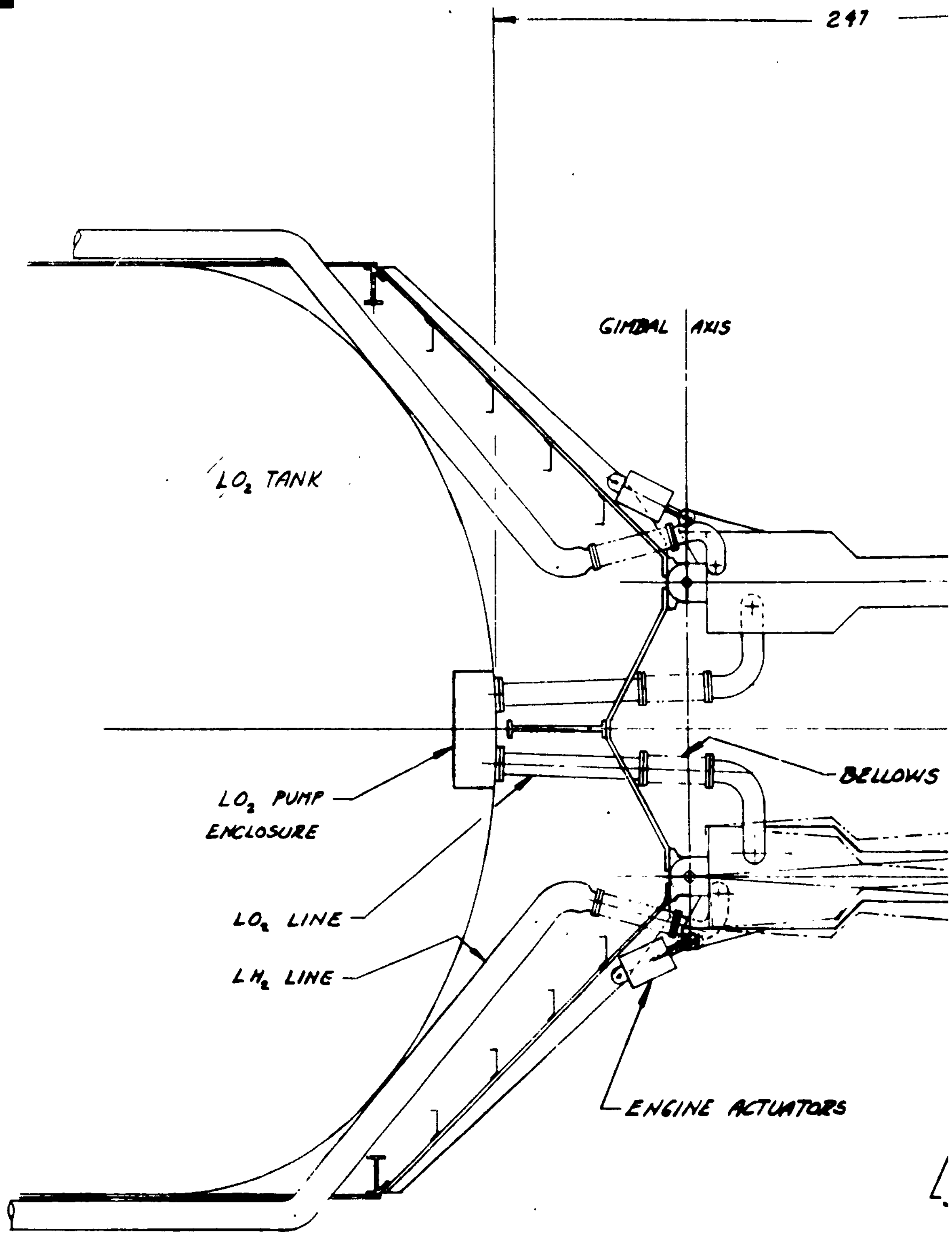
Table 8
Page 7 of 7

TOTALS PER VEHICLE

PAYLOAD 199434.

NUMBER OF VEHICLES	COST	COST-REC	COST/LB.PL	COST/LB.PL-REC
10.	1.804175E 08	0.	90.	0.
100.	9.770643E 08	0.	49.	0.
1000.	6.239786E 09	0.	31.	0.

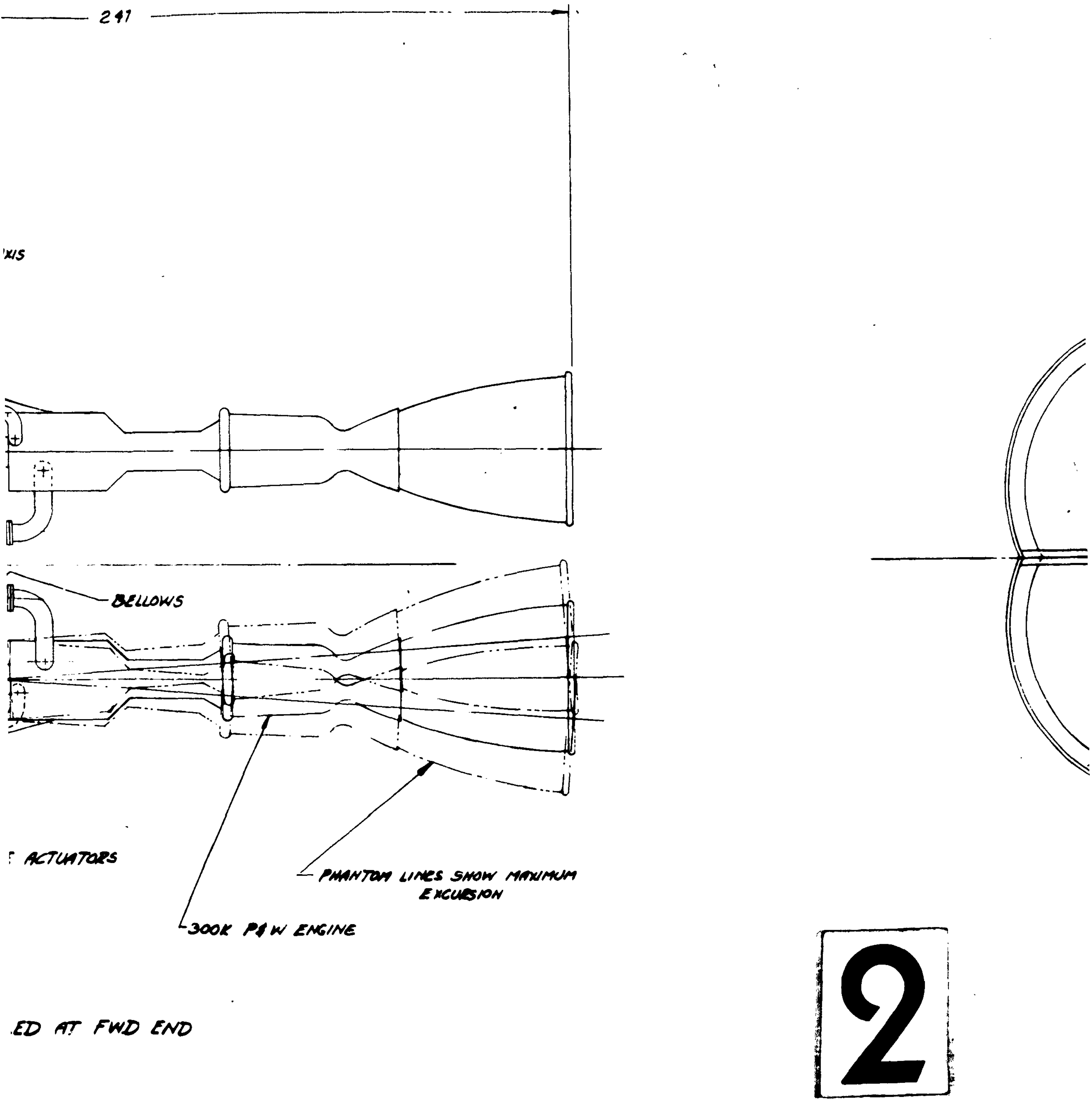
[redacted]



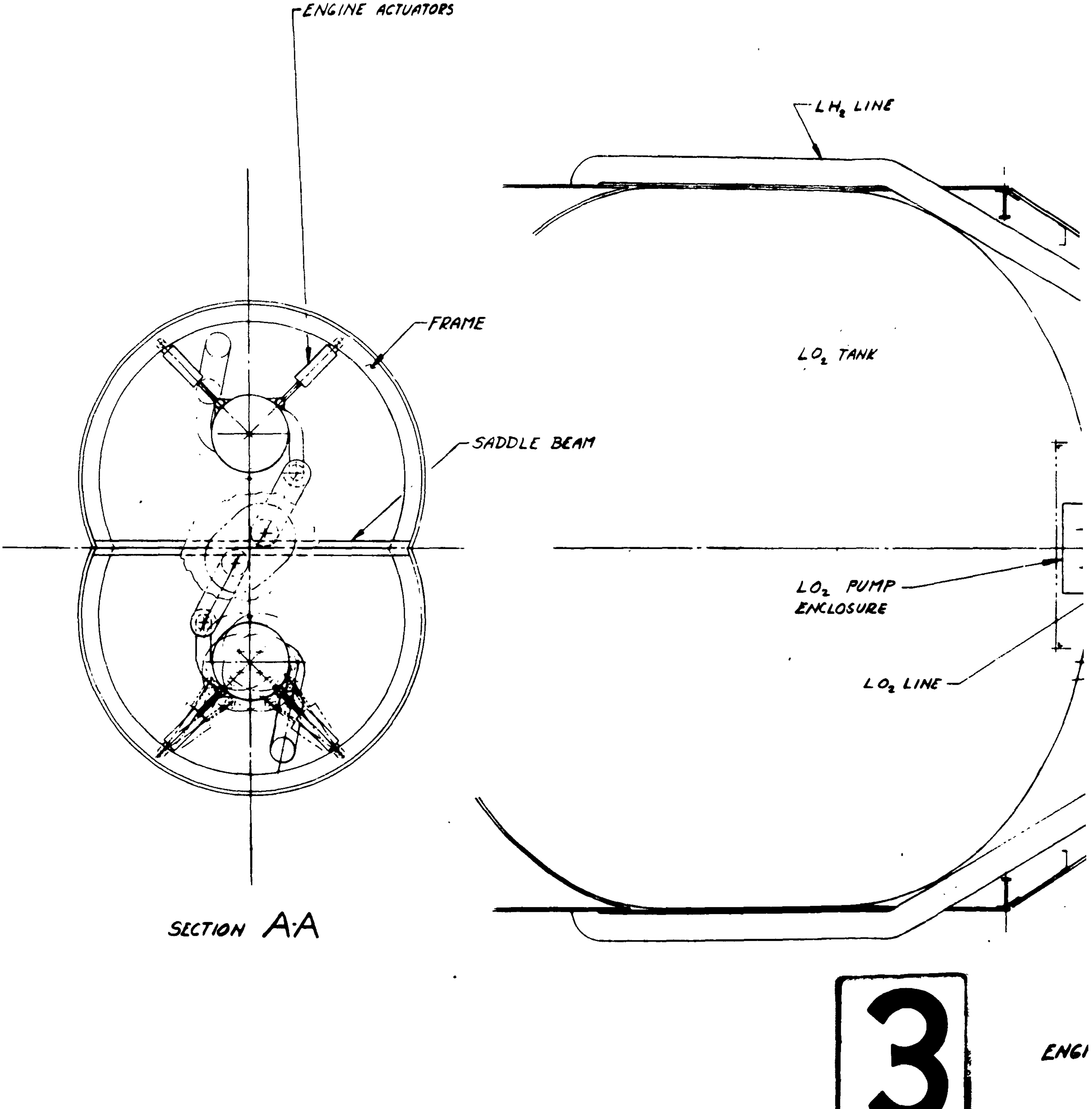
ENGINES GIMBALLED AT FWD END

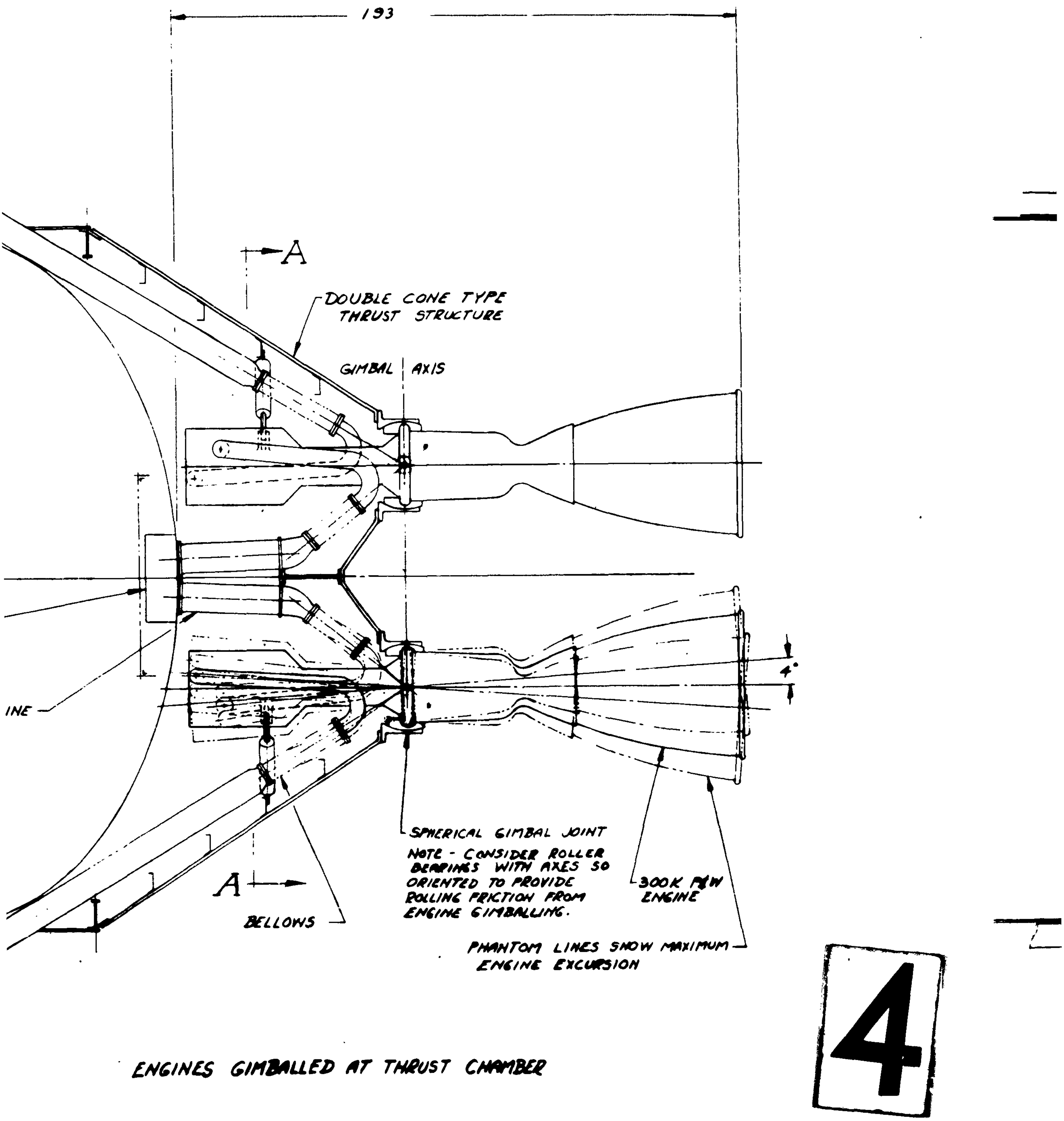
1

SID 61-341

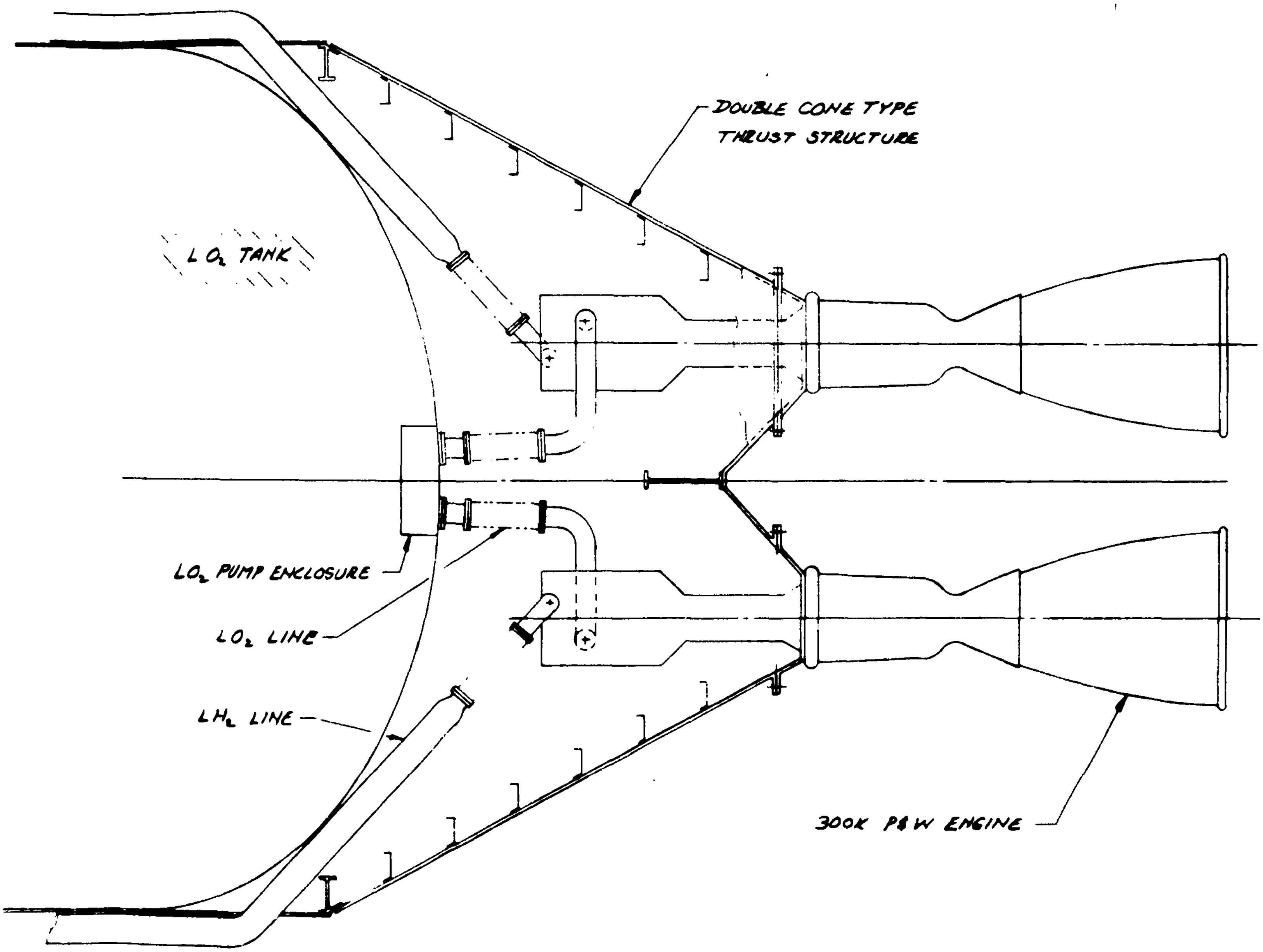


241
XIS
BELLOWS
ACTUATORS
PHANTOM LINES SHOW MAXIMUM EXCURSION
300K P&W ENGINE
ED AT FWD END
2


ENGINE ACTUATORS
FRAME
SADDLE BEAM
SECTION A-A
LH_2 LINE
LO_2 TANK
LO_2 PUMP ENCLOSURE
LO_2 LINE
ENGI



ENGINES GIMBALLED AT THRUST CHAMBER



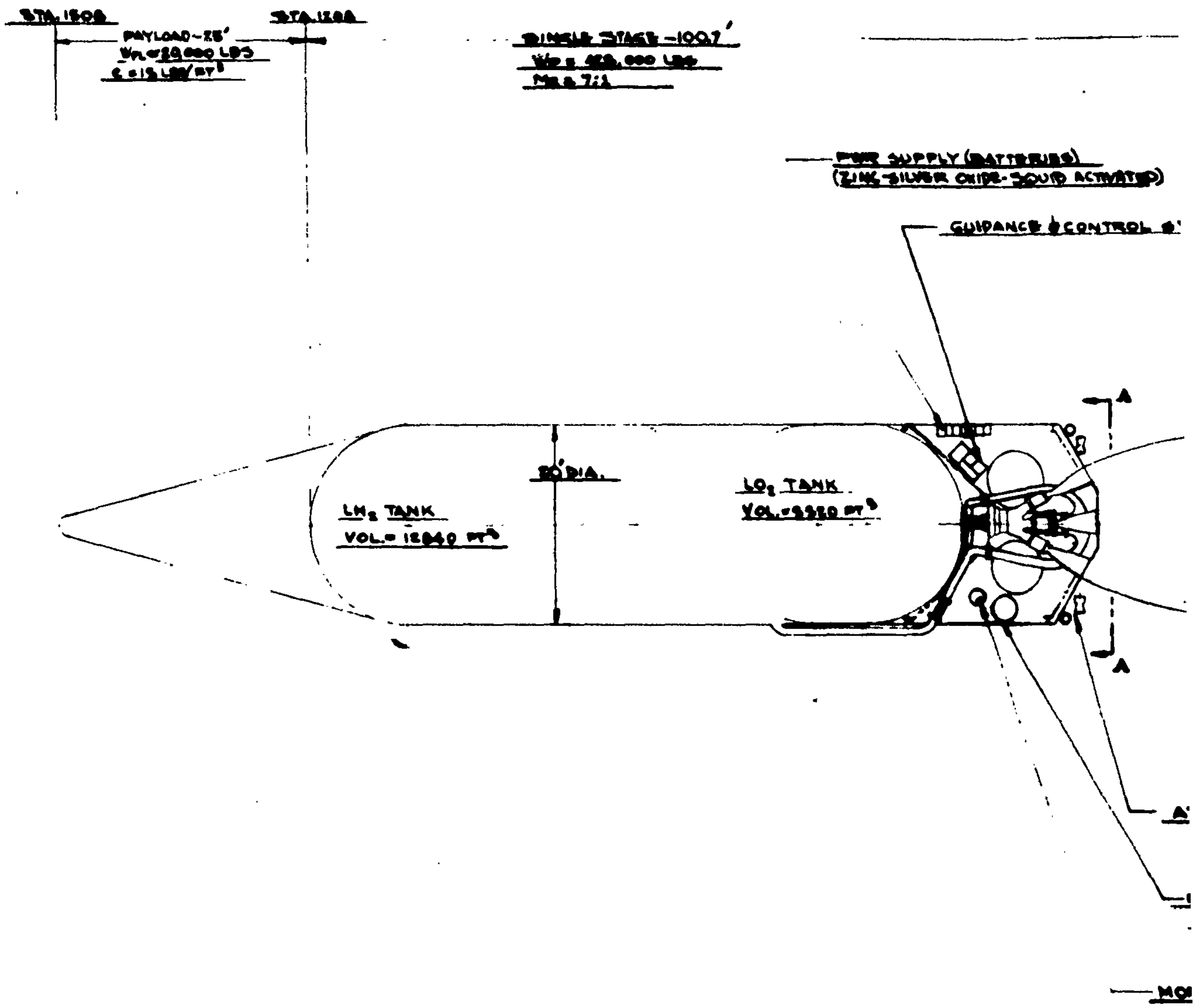
FIXED ENGINE INSTALLATION

5

Figure A-1. 300K Pratt & Whitney Engine Installation Engine Thrust Structure - Fixed and Gimbaled

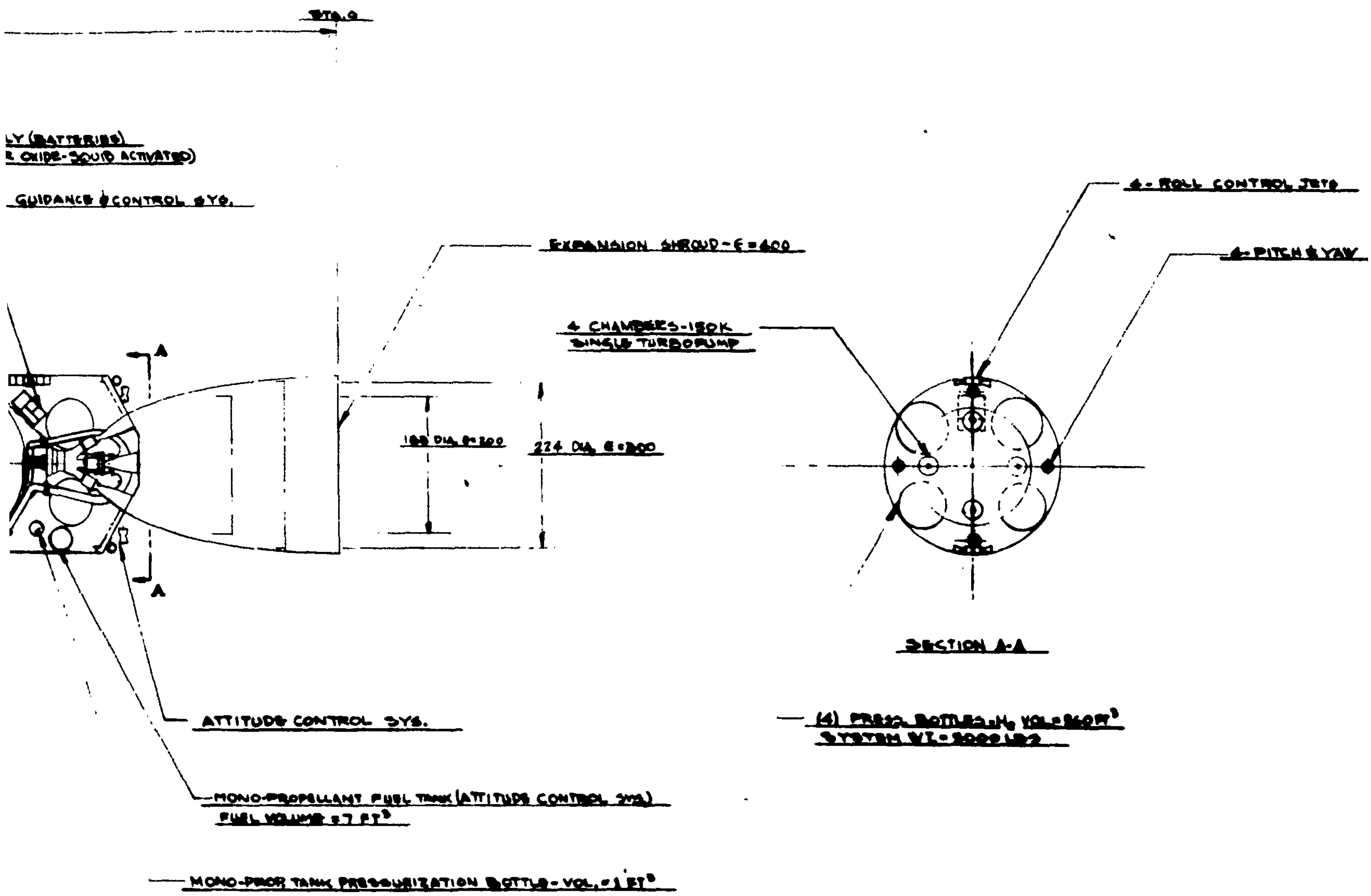
[redacted]

[redacted]



SID 61-341

1



2

Figure A-2. 600K Pratt & Whitney Engines-Single-Sta

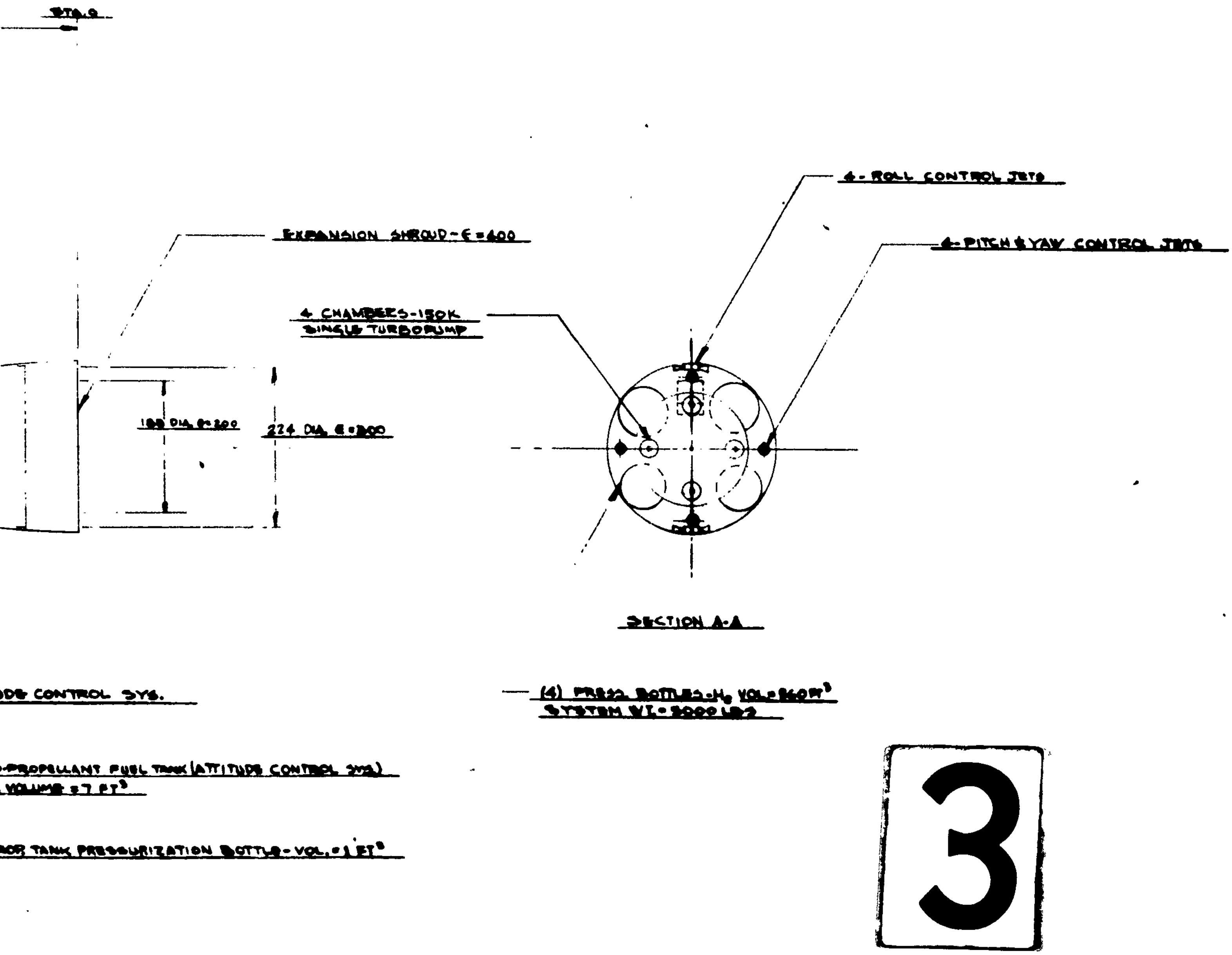
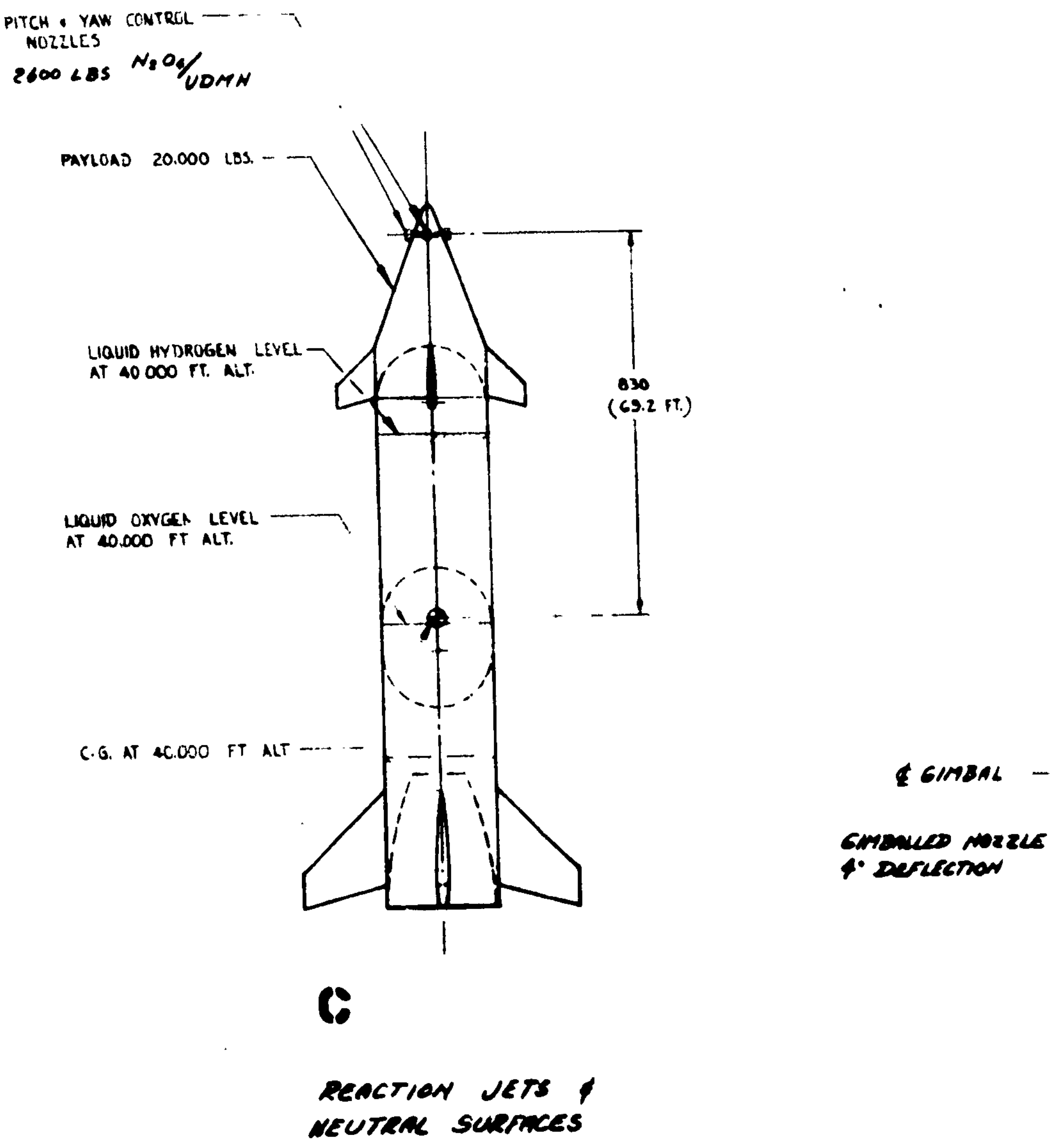
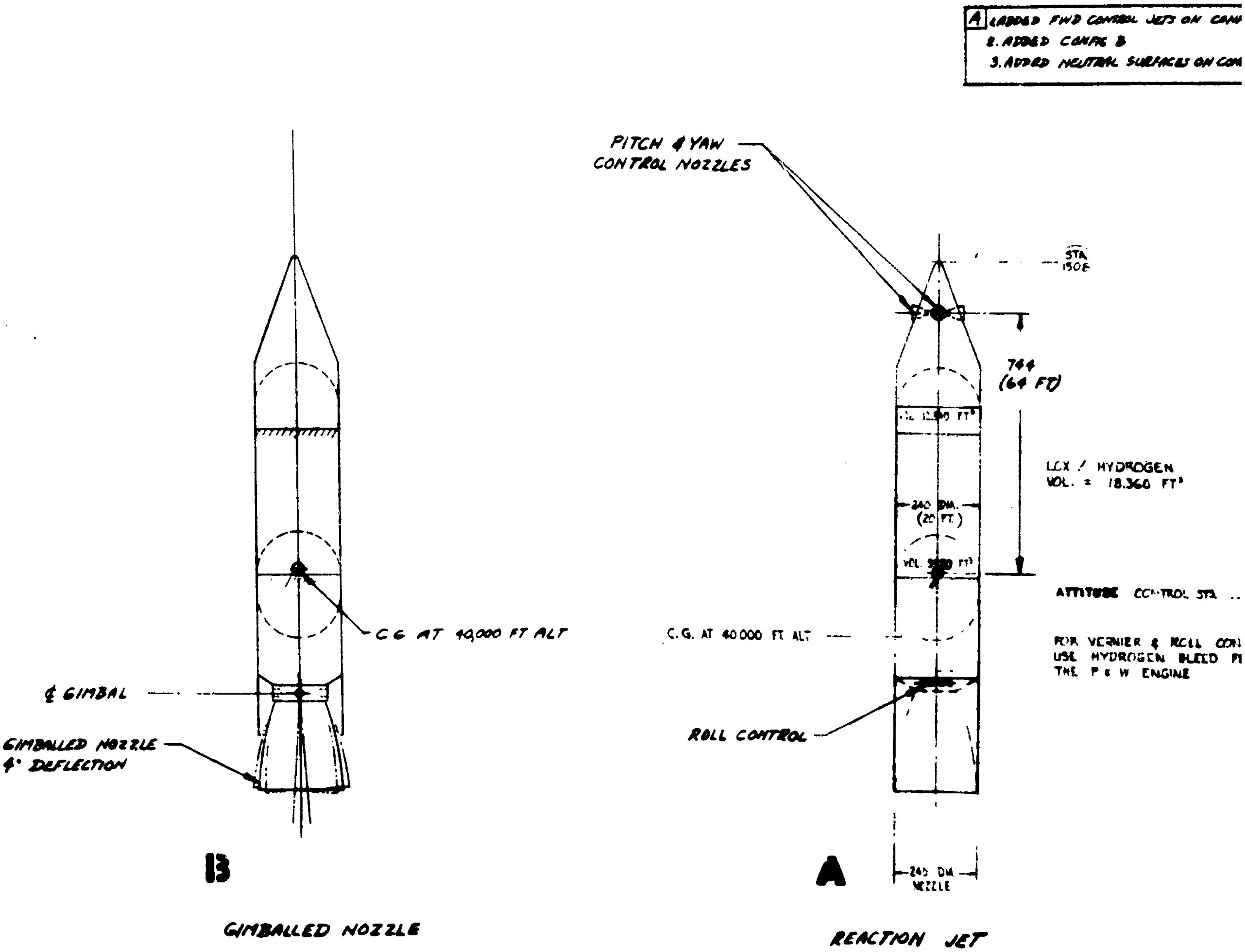


Figure A-2. 600K Pratt & Whitney Engines-Single-Stage-to-Orbit

[redacted]


PITCH & YAW CONTROL NOZZLES
2600 LBS N_2O_4/UDMH
PAYLOAD 20.000 LBS.
LIQUID HYDROGEN LEVEL AT 40 000 FT. ALT.
830 (69.2 FT.)
LIQUID OXYGEN LEVEL AT 40.000 FT ALT.
C.G. AT 40.000 FT ALT
¢ GIMBAL
GIMBALLED NOZZLE 4° DEFLECTION
C
REACTION JETS & NEUTRAL SURFACES

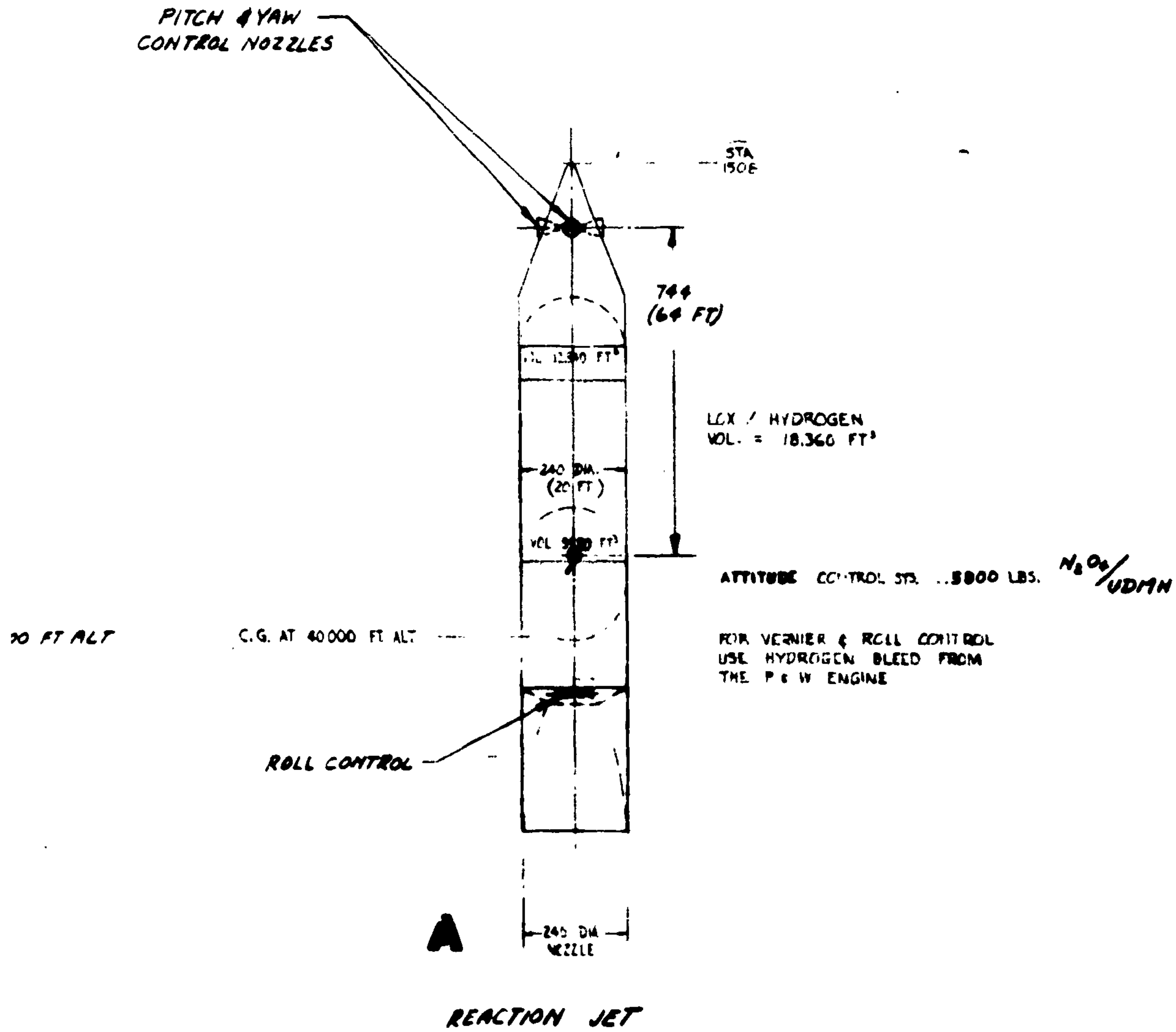
1



2

Figure A-3. Stability Compa

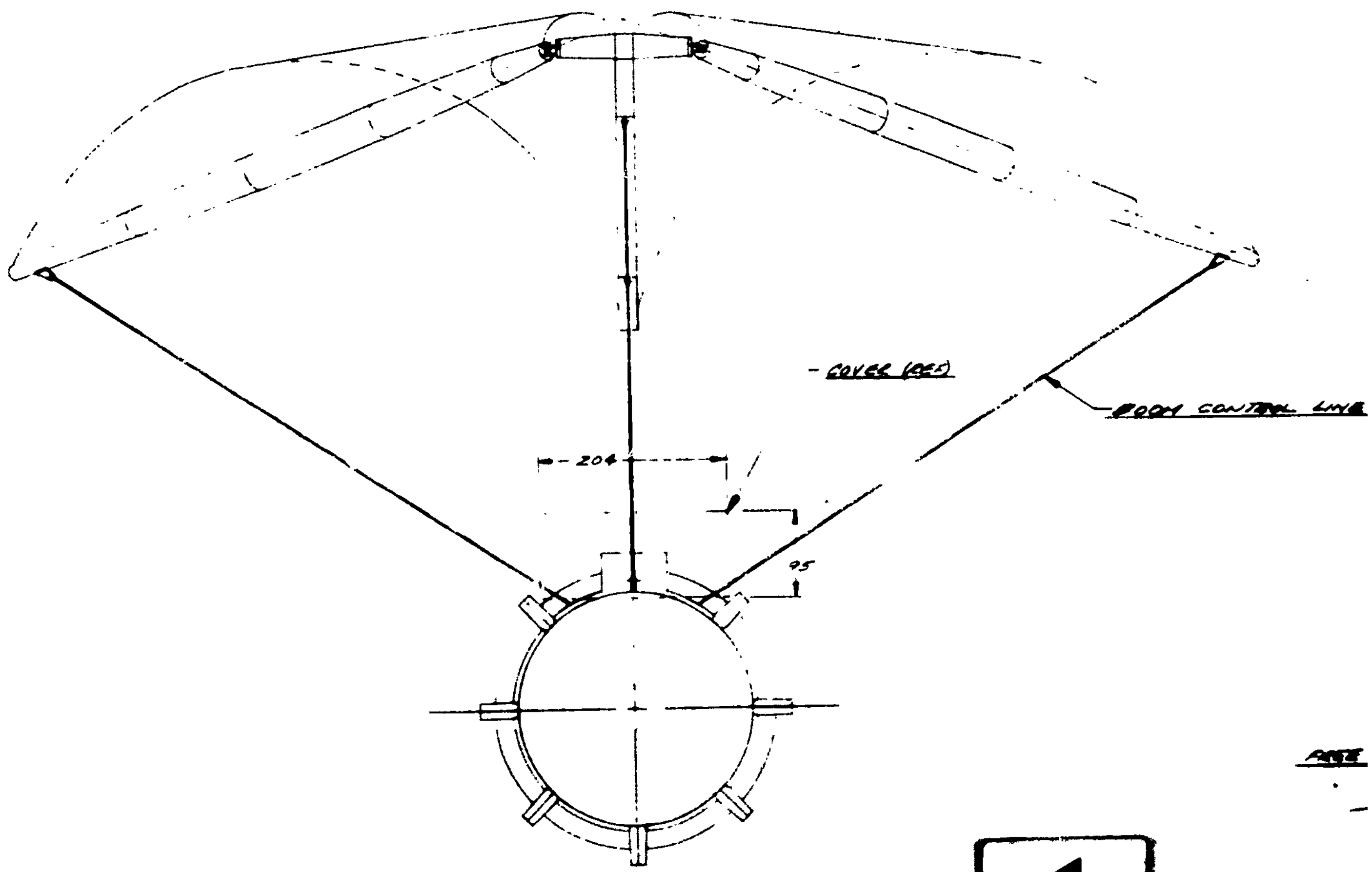
A 1. ADDED FWD CONTROL JETS ON CONFIG. A 2. ADDED CONFIG B 3. ADDED NEUTRAL SURFACES ON CONFIG C	DAILEY 8-28-61



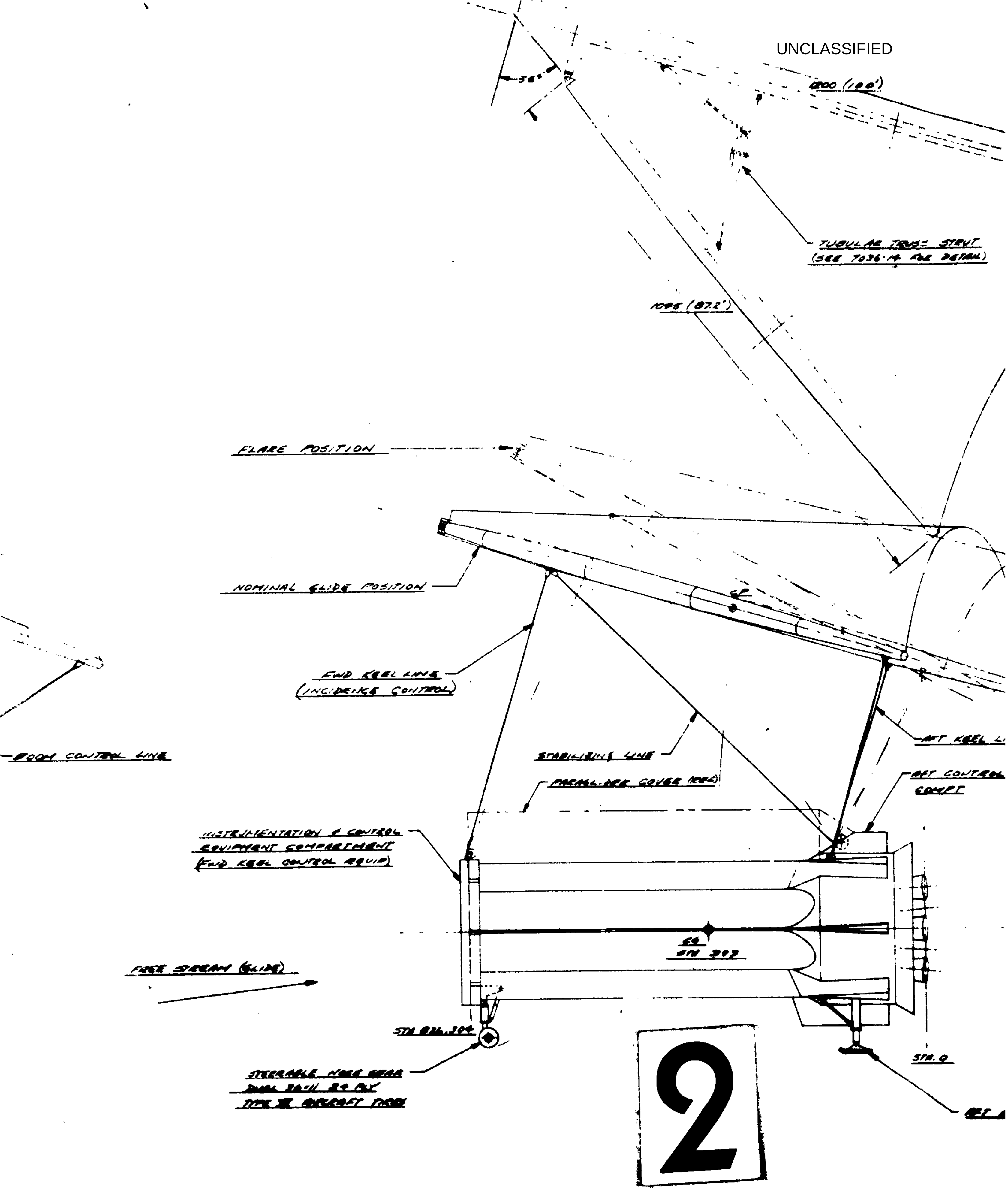
3

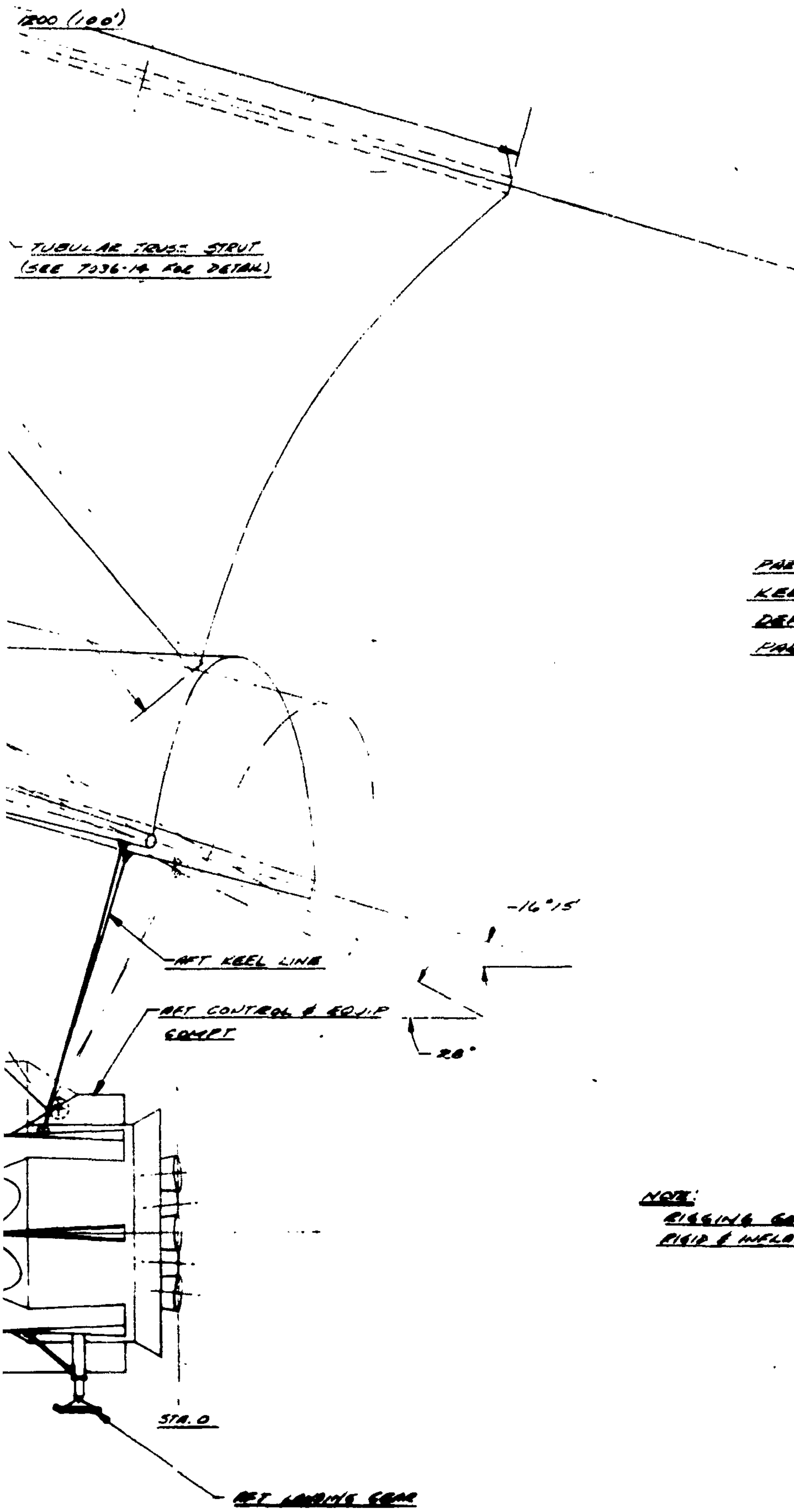
Figure A-3. Stability Comparison Single-Stage-to-Orbit

[redacted]



1


1200 (100')
TUBULAR TRUSS STRUT
(SEE 7036-14 FOR DETAIL)
1095 (87.2')
FLARE POSITION
NOMINAL GLIDE POSITION
FWD KEEL LINE
(INCIDENCE CONTROL)
BOOM CONTROL LINE
STABILIZING LINE
PARAGLIDER COVER (REF)
INSTRUMENTATION & CONTROL
EQUIPMENT COMPARTMENT
(FWD KEEL CONTROL EQUIP)
CG
STA 393
FREE STREAM (GLIDE)
STA 836.304
STEERABLE NOSE GEAR
DUAL 26x11 24 PLY
TYPE III AIRCRAFT TIRES
STA. 0
2

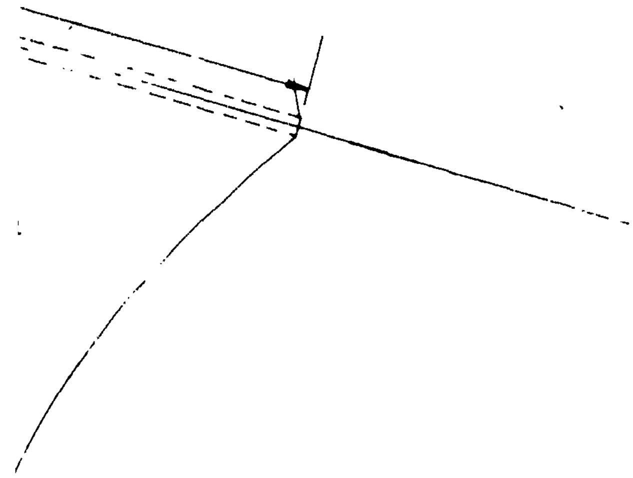


PARAGLIDER FLAT PATTERN AREA	7070 FT^2
KEEL & BOOM LENGTHS (NOMINAL)	100 FT
DEPLOYED SWEEP ANGLE	55°
PARAGLIDER ATTACH CABLE LENGTHS	
FWD KEEL NOMINAL GLIDE	44.2 FT
PRE-FLARE	36.7 FT
FLARE	55 FT
AFT KEEL	27.5 FT
STABILIZING LINE	50.8 FT
BOOM LINES	50.5 FT

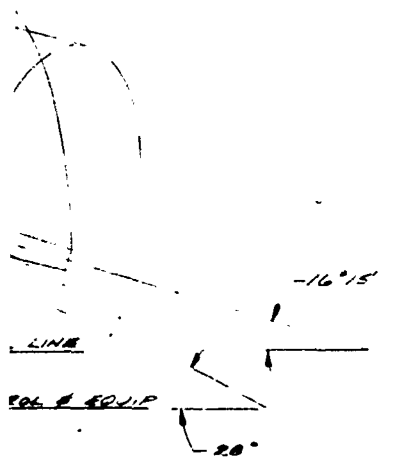
3

NOTE:
RIGGING GEOMETRY TYP FOR
RIGID & INFLATED MEMBER KITES

Figure A-4. Rigid Boom Paraglider



PARAGLIDER FLAT PATTERN AREA	7070 FT^2
KEEL & BOOM LENGTHS (NOMINAL)	100 FT
DEPLOYED SWEEP ANGLE	55°
PARAGLIDER ATTACH CABLE LENGTHS	
FWD KEEL NOMINAL GLIDE	44.2 FT
PRE-FLARE	36.7 FT
FLARE	55 FT
AFT KEEL	27.5 FT
STABILIZING LINE	50.3 FT
BOOM LINES	50.5 FT



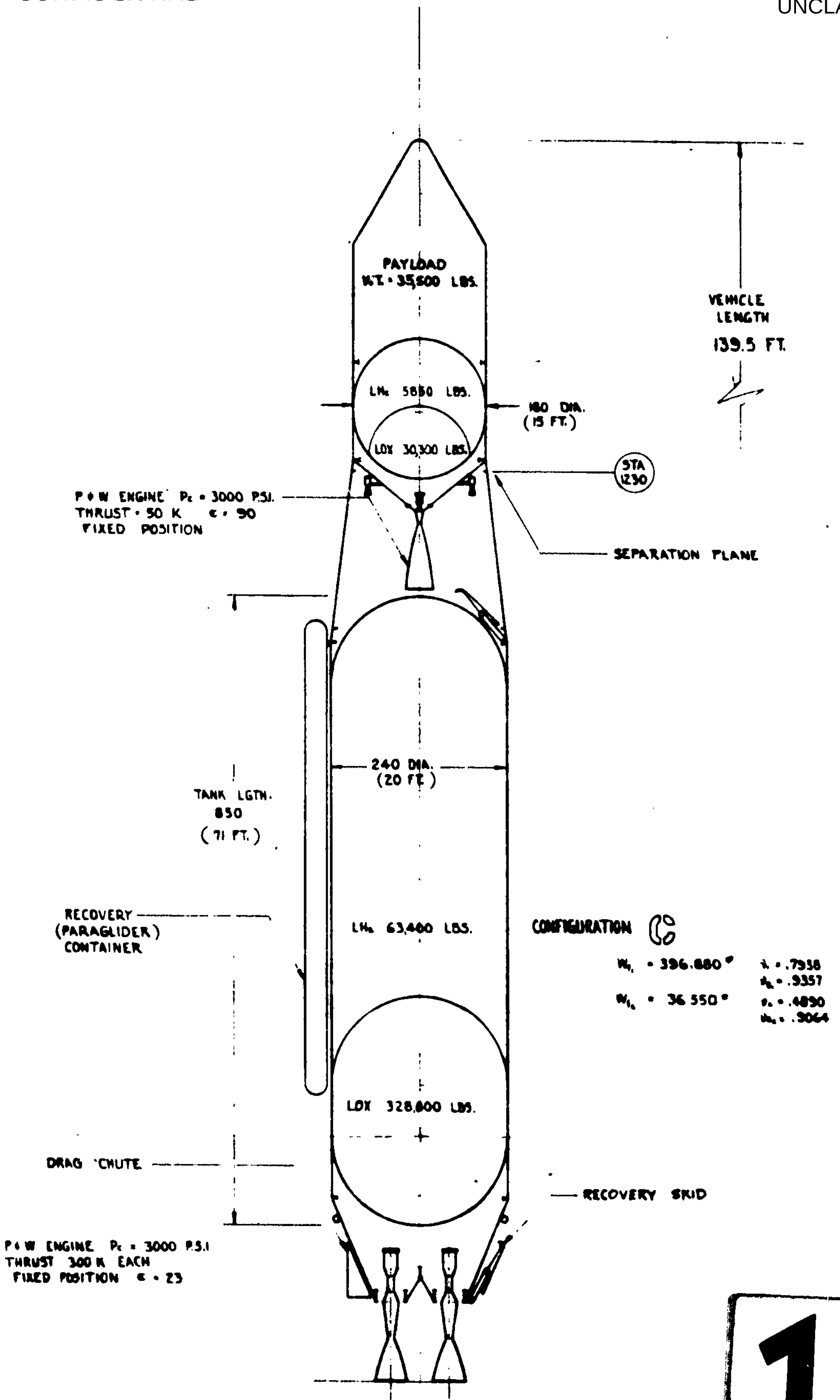
NOTE:
RIGGING GEOMETRY TYP FOR
RIGID & INFLATED MEMBER KITES

4

Figure A-4. Rigid Boom Paraglider

T LANDING GEAR

[redacted]


PAYLOAD
WT. = 35,500 LBS.
VEHICLE
LENGTH
139.5 FT.
LH₂ 5850 LBS.
LOX 30,300 LBS.
160 DIA.
(15 FT.)
STA
1230
P & W ENGINE Pc = 3000 PSI.
THRUST = 50 K ε = 90
FIXED POSITION
SEPARATION PLANE
240 DIA.
(20 FT.)
TANK LGTH.
850
(71 FT.)
RECOVERY
(PARAGLIDER)
CONTAINER
LH₂ 63,400 LBS.
CONFIGURATION
= 396,880 #
= 36,550 #
= .7938
= .9357
= .4890
= .3064
LOX 328,800 LBS.
DRAG CHUTE
RECOVERY SKID
P & W ENGINE Pc = 3000 P.S.I
THRUST 300 K EACH
FIXED POSITION ε = 23

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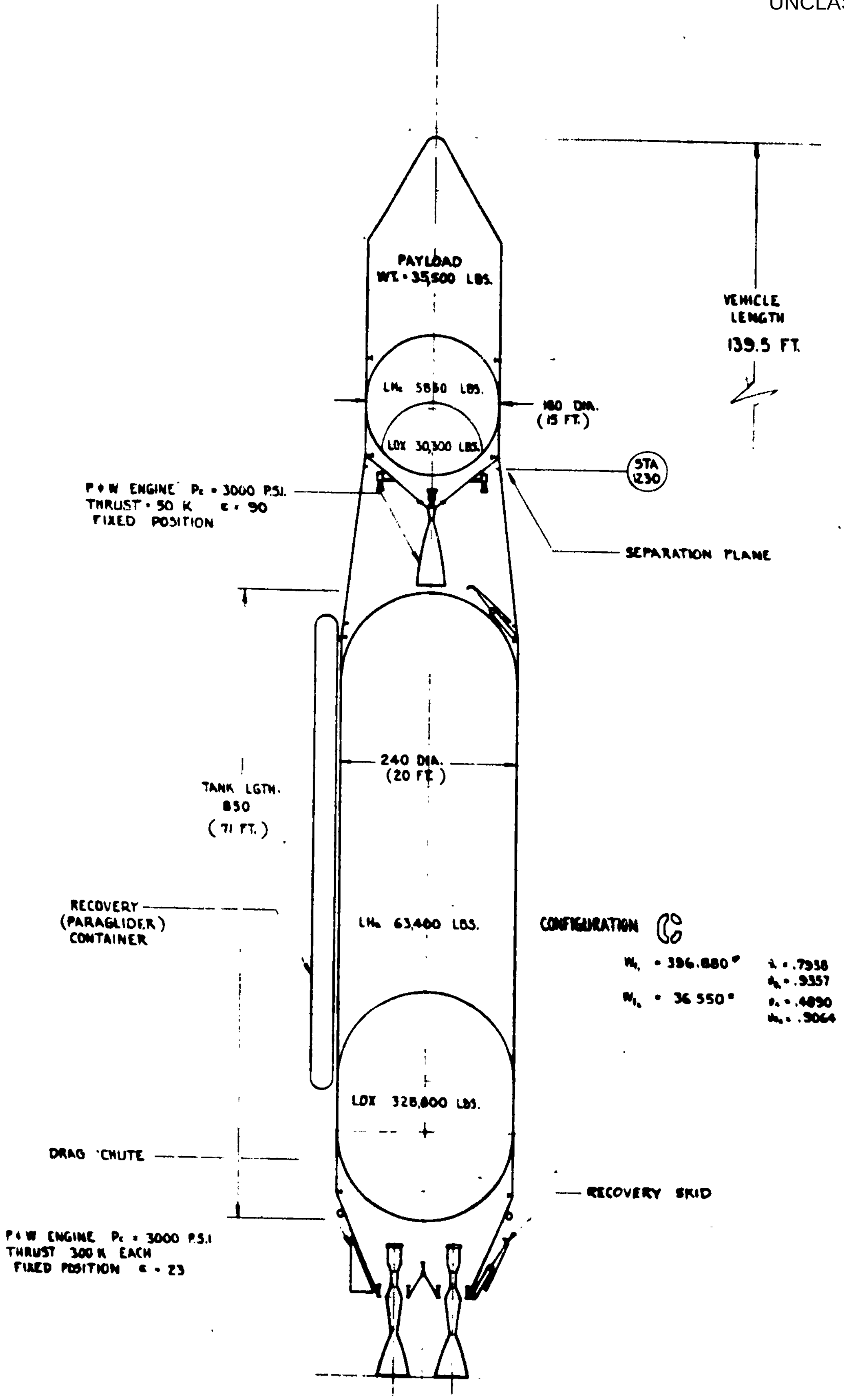
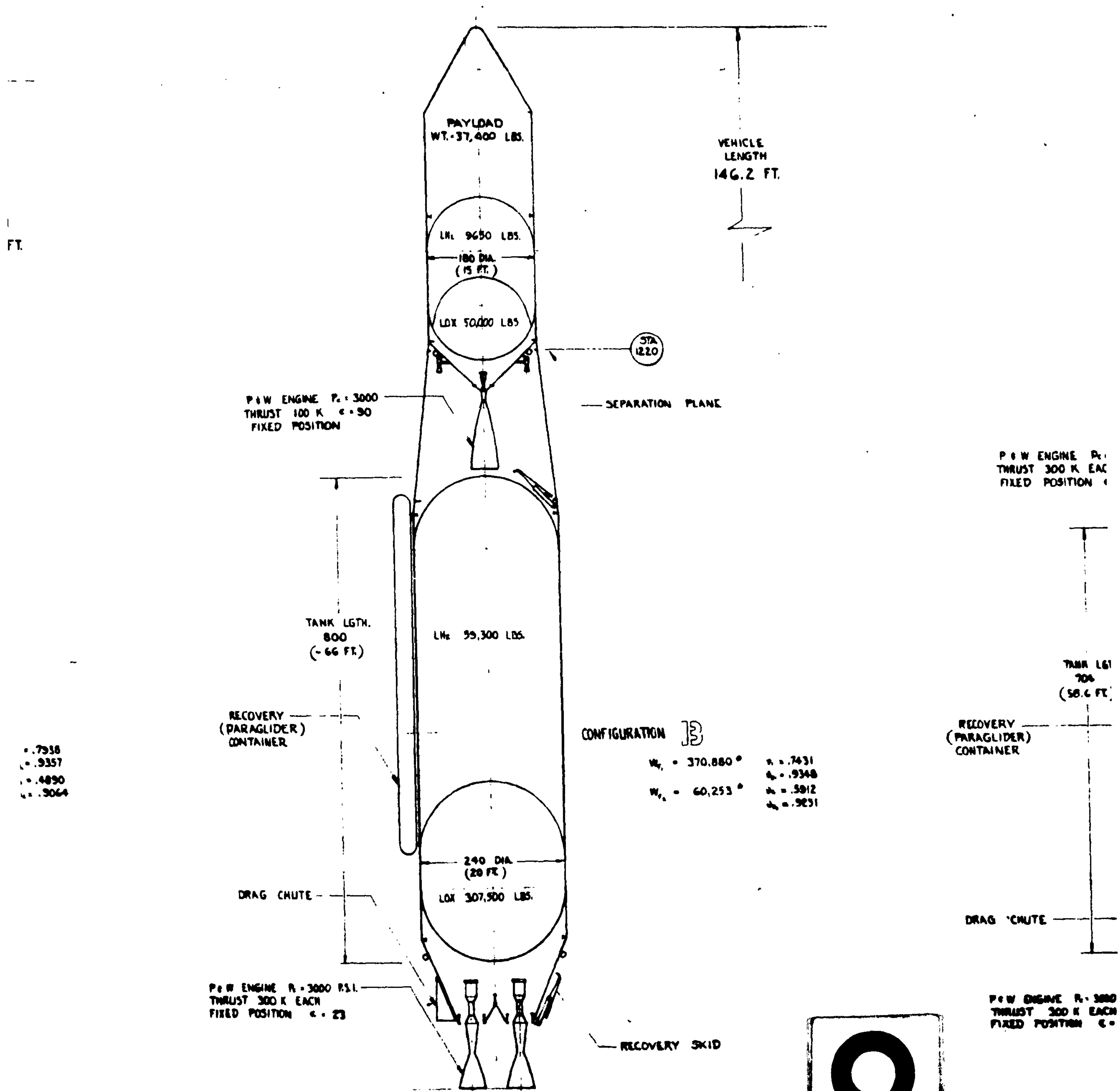
1

CONFIDENTIAL



SID 61-341

1



PAYLOAD
WT.-37,400 LBS.
VEHICLE LENGTH
146.2 FT.
LH2 9650 LBS.
180 DIA. (15 FT.)
LOX 50,000 LBS
STA 1220
P & W ENGINE Pc = 3000
THRUST 100 K ε = 90
FIXED POSITION
SEPARATION PLANE
TANK LGTH. 800 (~66 FT.)
LH2 59,300 LBS.
RECOVERY (PARAGLIDER) CONTAINER
CONFIGURATION B
240 DIA. (20 FT.)
LOX 307,500 LBS.
DRAG CHUTE
P & W ENGINE Pc = 3000 P.S.I.
THRUST 300 K EACH
FIXED POSITION ε = 23
RECOVERY SKID

2

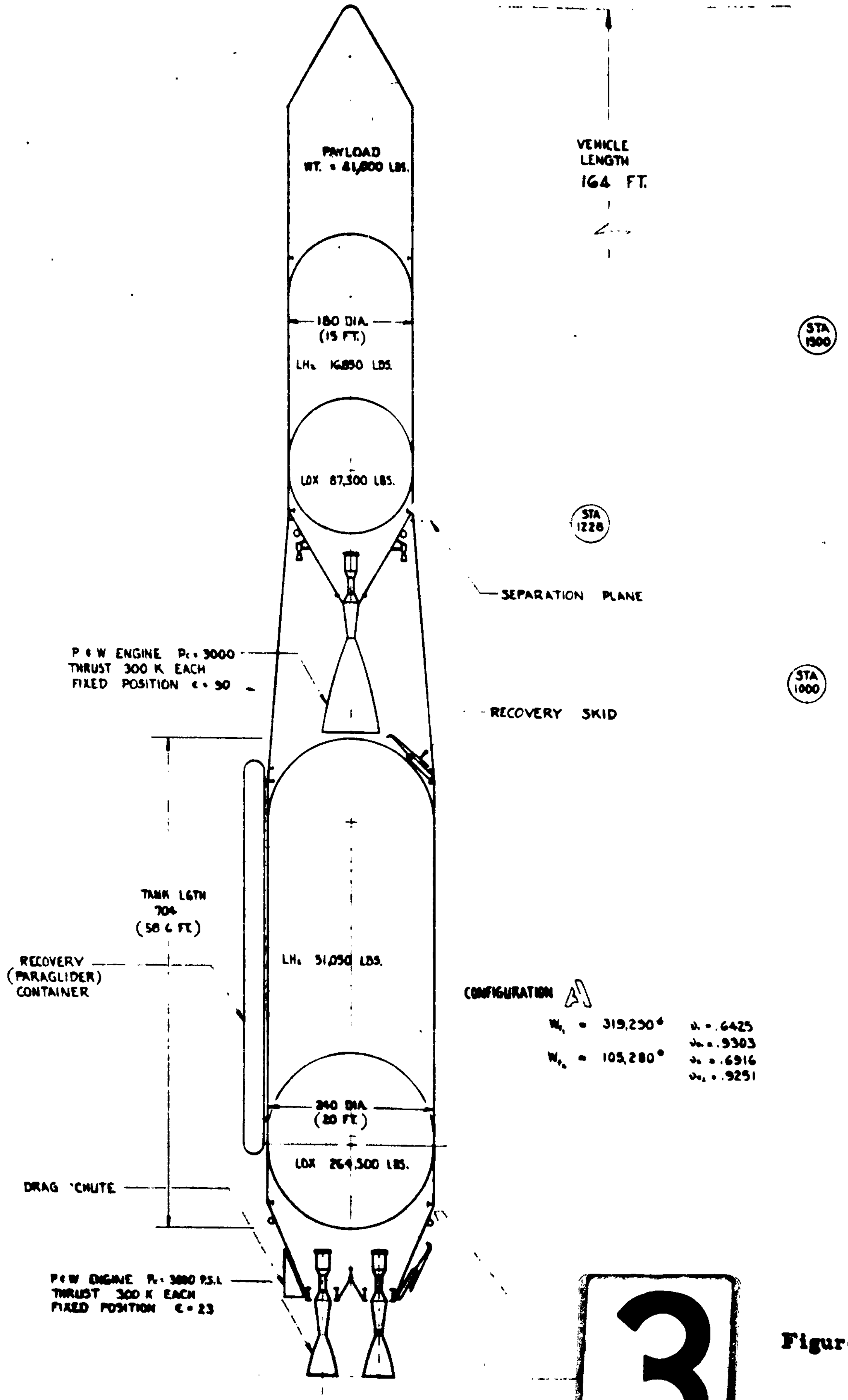
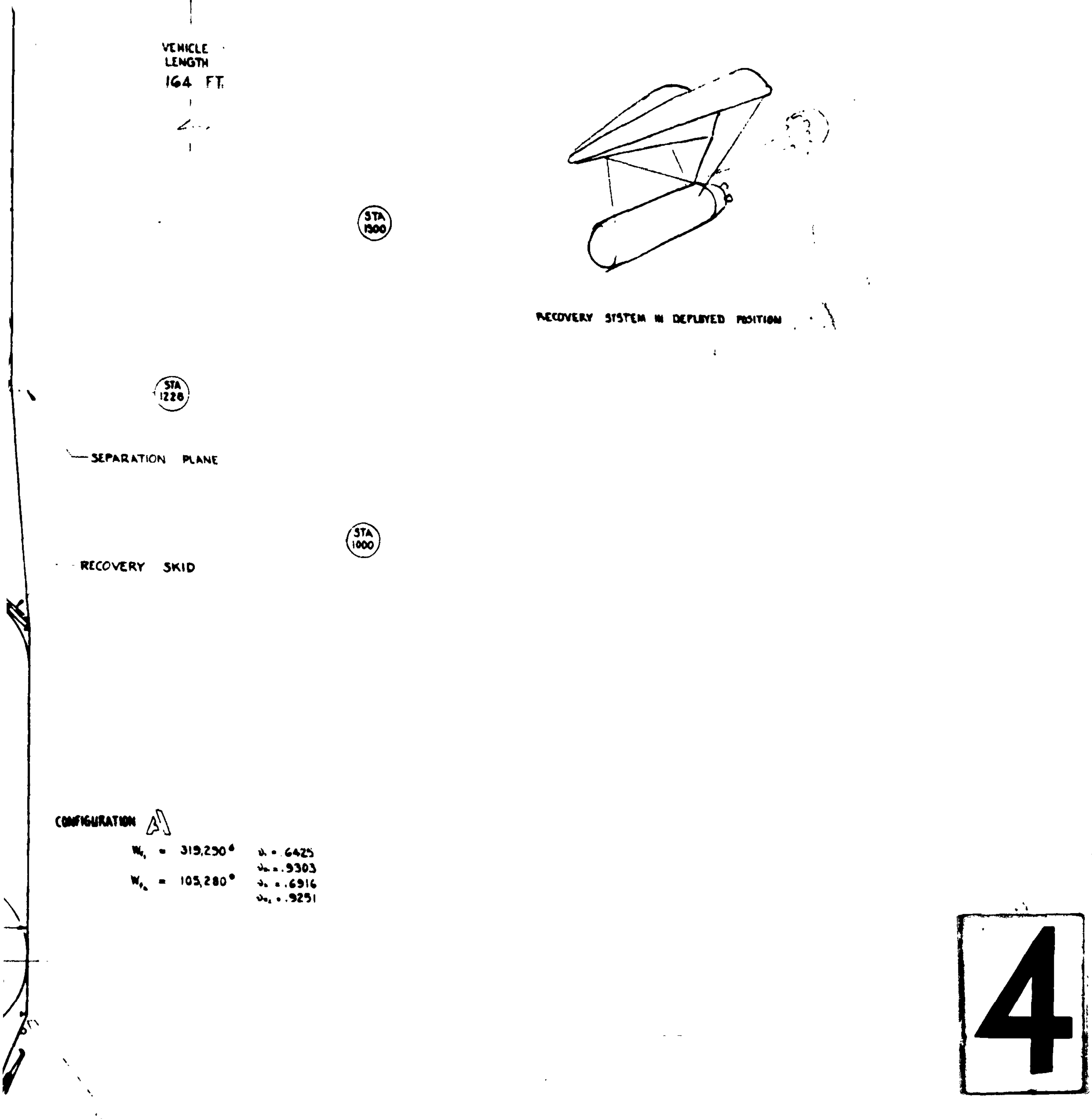


Figure A-5. Advanced Pratt an



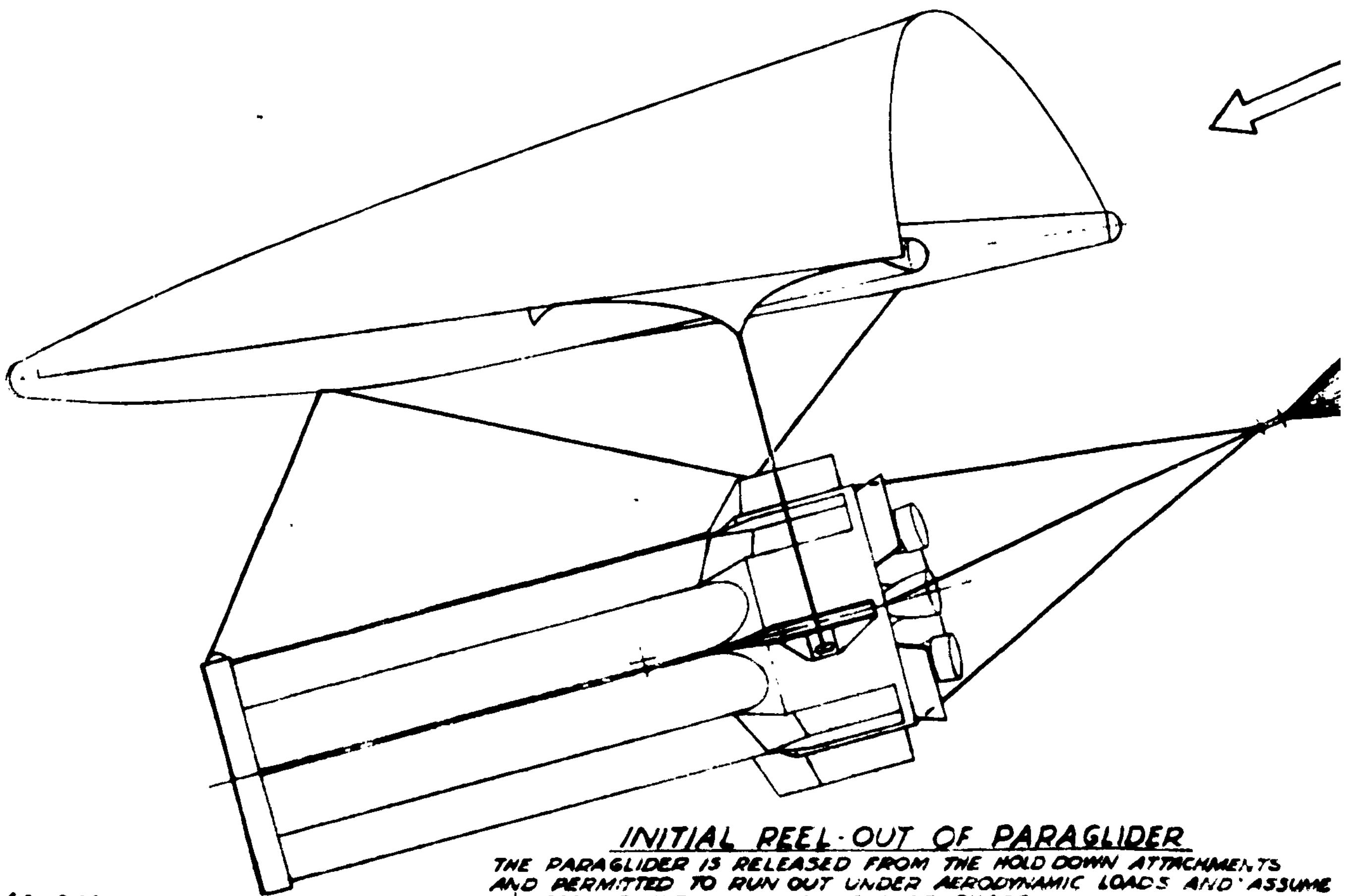
4

Figure A-5. Advanced Pratt and Whitney-Paraglider Recovery

[redacted]

[redacted]

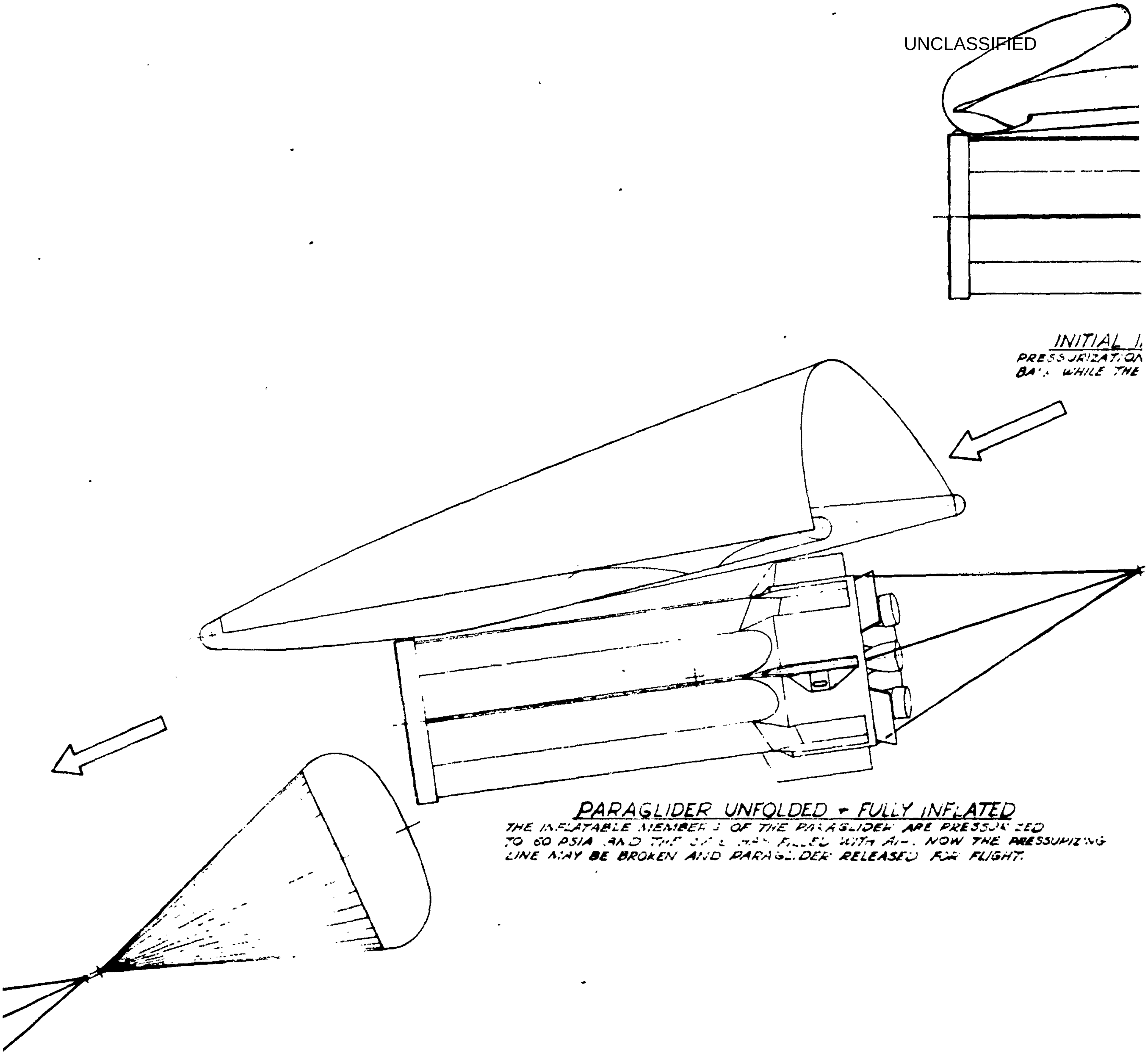
1



INITIAL REEL-OUT OF PARAGLIDER

THE PARAGLIDER IS RELEASED FROM THE HOLD DOWN ATTACHMENTS AND PERMITTED TO RUN OUT UNDER AERODYNAMIC LOADS AND ASSUME IT'S PROPER ATTITUDE FOR THE GLIDE PHASE

SID 61-341

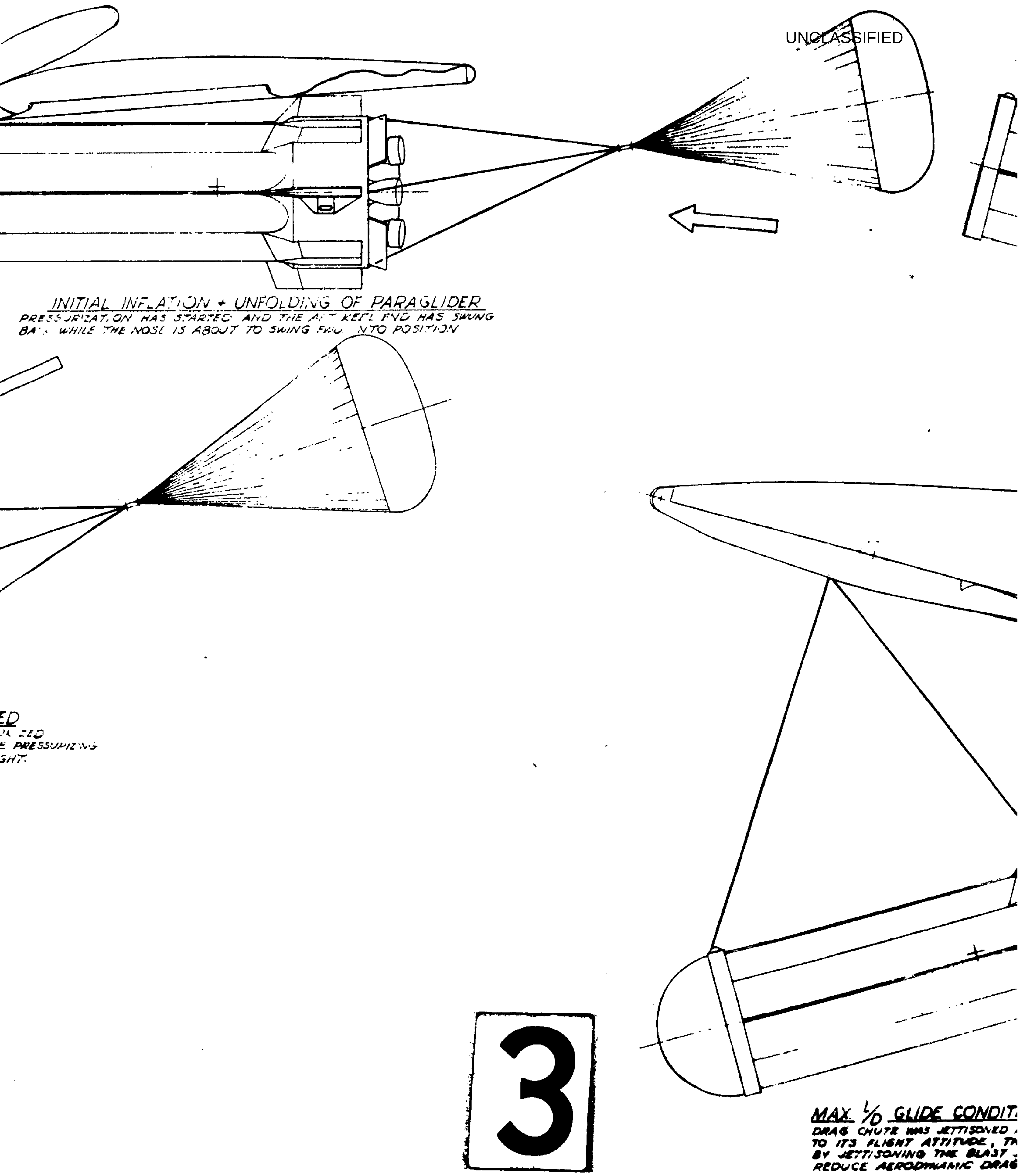


PARAGLIDER UNFOLDED + FULLY INFLATED

THE INFLATABLE MEMBERS OF THE PARAGLIDER ARE PRESSURIZED TO 60 PSIA AND THE SAIL HAS FILLED WITH AIR. NOW THE PRESSURIZING LINE MAY BE BROKEN AND PARAGLIDER RELEASED FOR FLIGHT.

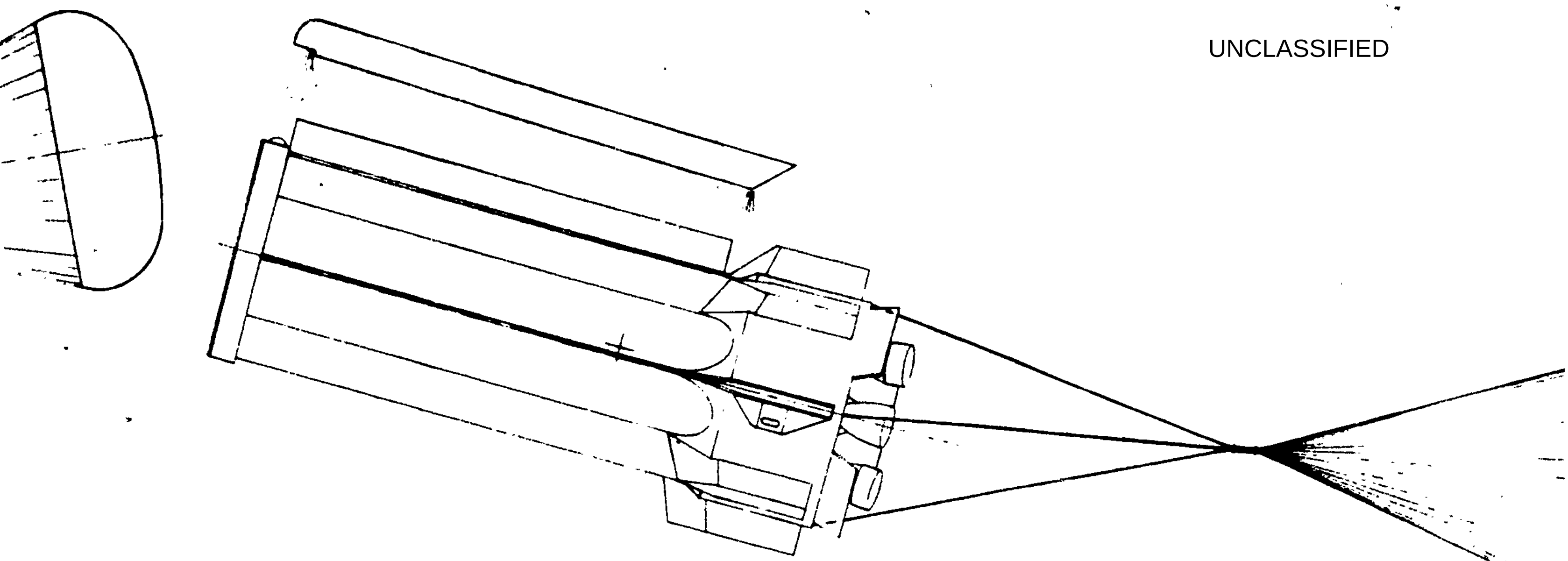
ER
TTACHMENTS
DS AND ASSUME

2



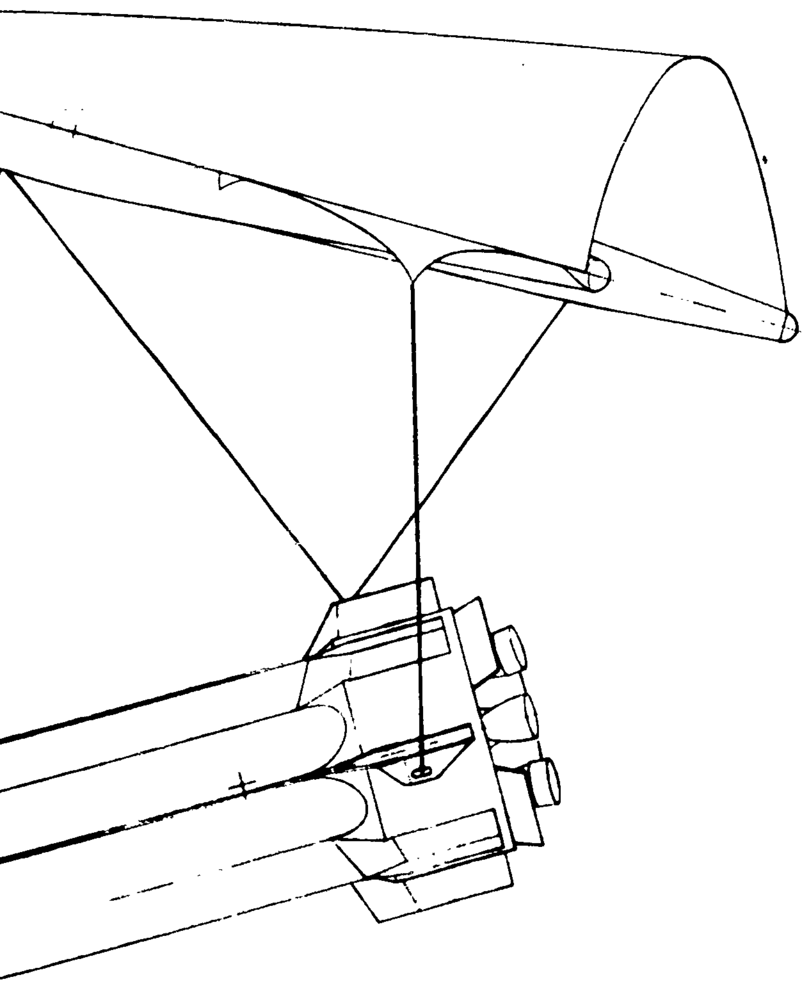
3

 Distribution A: Approved for public release; distribution unlimited. PA Case #10460.



JETTISON OF PARAGLIDER CANISTER

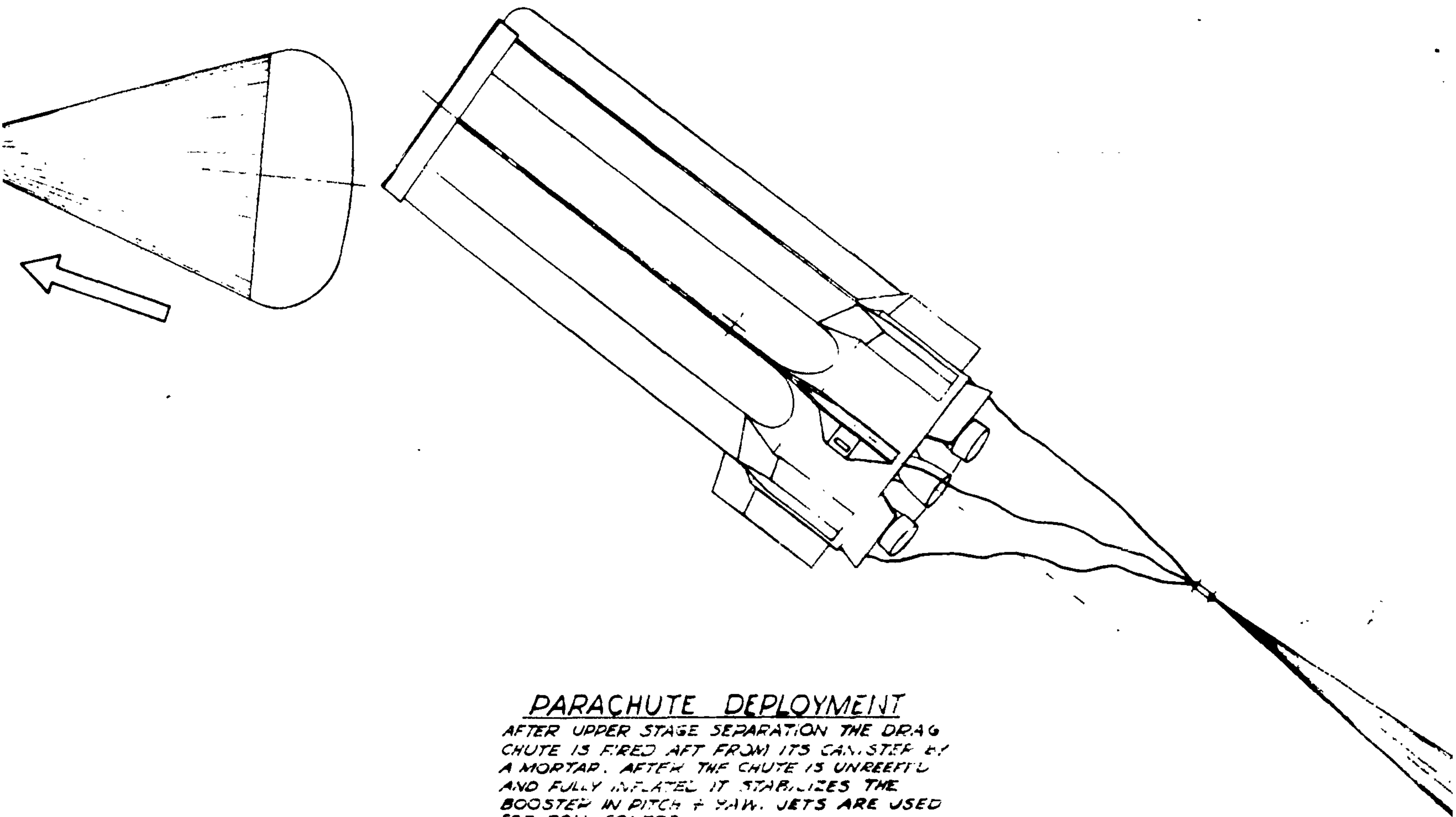
SOLID PROPELLANT ROCKETS PUSH THE CANISTER UP + AWAY FROM THE STOWED PARAGLIDER



AX. L/D GLIDE CONDITION + INFLATED NOSE COVER

AG CHUTE WAS JETTISONED AFTER THE PARAGLIDER WAS RUN OUT ITS FLIGHT ATTITUDE, THE NOSE FAIRING HAS BEEN UNCOVERED JETTISONING THE BLAST SHIELD AND THE FAIRING INFLATED TO RODUCE AERODYNAMIC DRAG

4

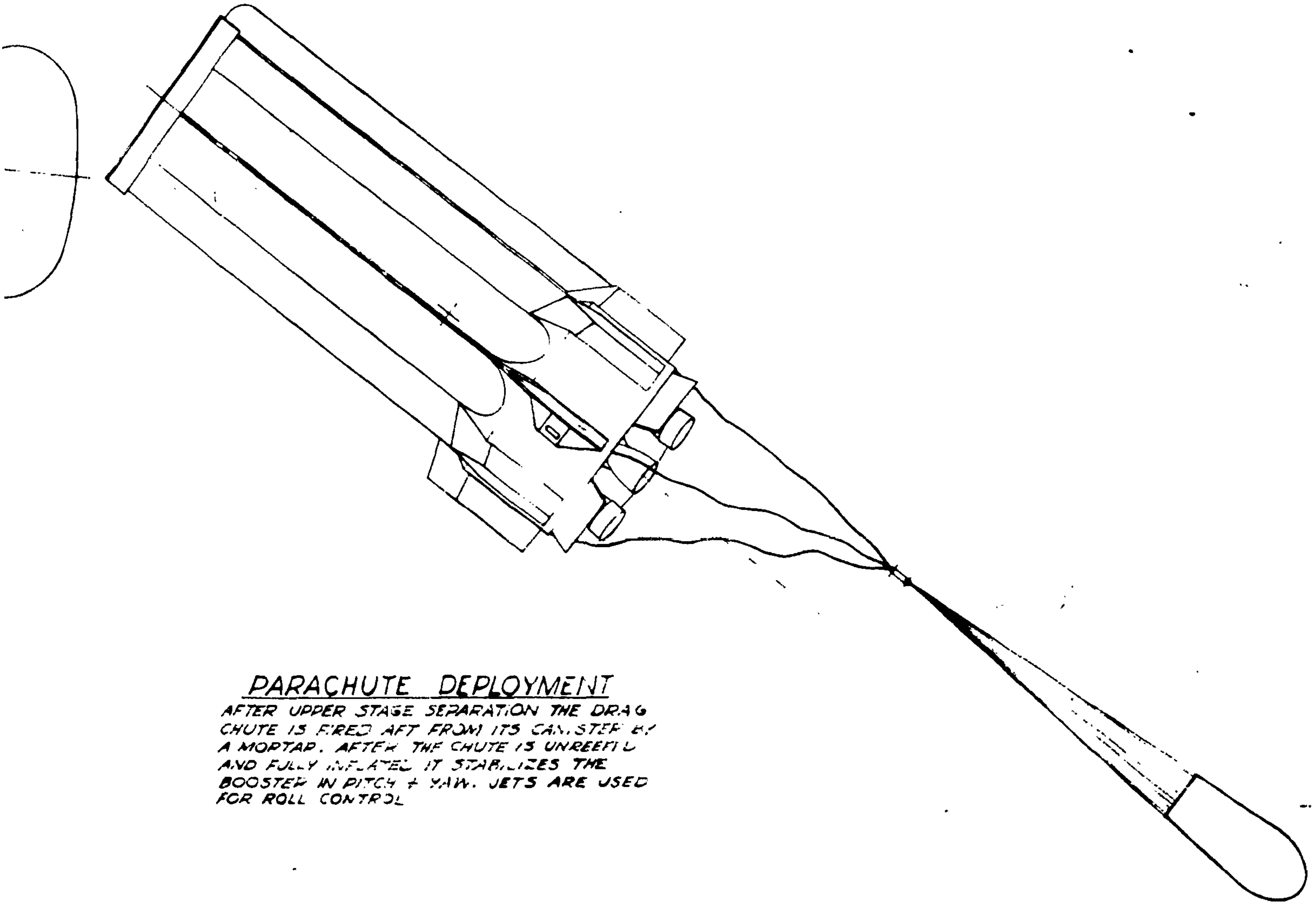


PARACHUTE DEPLOYMENT

AFTER UPPER STAGE SEPARATION THE DRAG CHUTE IS FIRED AFT FROM ITS CANISTER BY A MORTAR. AFTER THE CHUTE IS UNREEFED AND FULLY INFLATED IT STABILIZES THE BOOSTER IN PITCH + YAW. JETS ARE USED FOR ROLL CONTROL

5

Figure A-6. Inflatable Paraglider

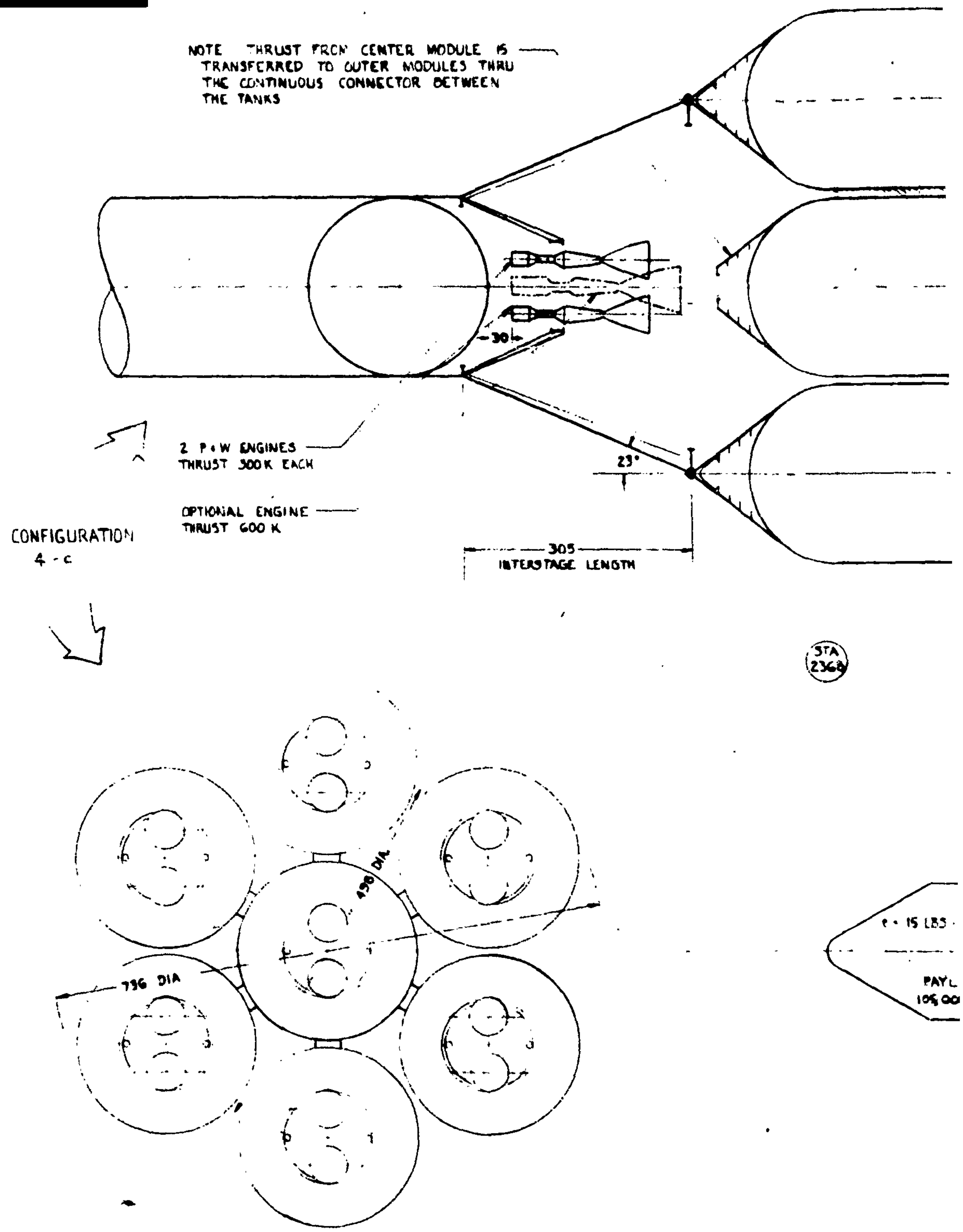


6

Figure A-6. Inflatable Paraglider

[redacted]

[redacted]

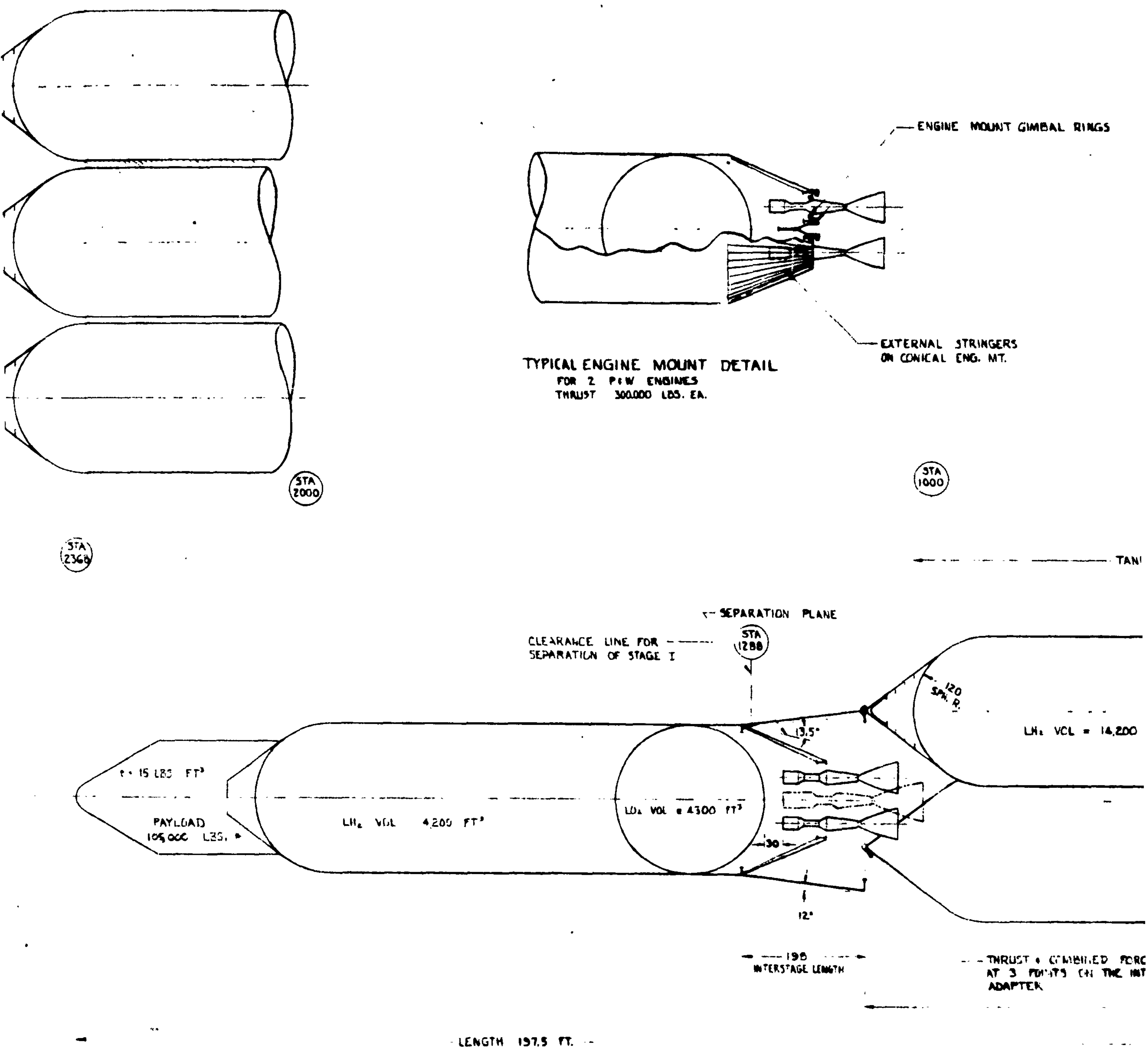


REAR VIEW

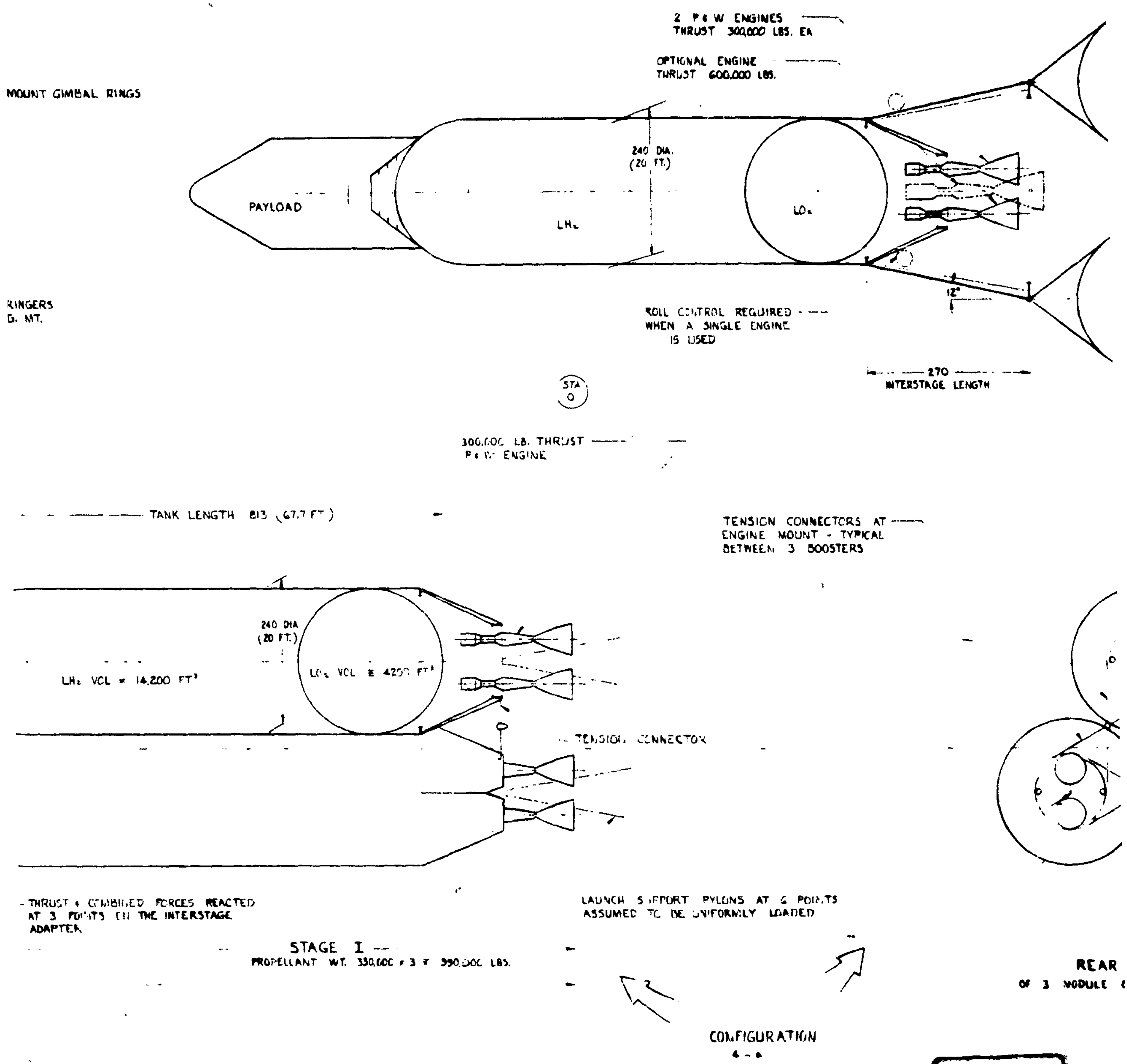
OF 7 MODULE CLUSTER OF TANKS

SID 61-341

1


ENGINE MOUNT GIMBAL RINGS
EXTERNAL STRINGERS
ON CONICAL ENG. MT.
TYPICAL ENGINE MOUNT DETAIL
FOR 2 P&W ENGINES
THRUST 300,000 LBS. EA.
STA 2000
STA 1000
STA 2368
TANK
SEPARATION PLANE
CLEARANCE LINE FOR
SEPARATION OF STAGE I
STA 1288
120
SPH. R.
LH₂ VOL = 14,200
13.5°
PAYLOAD
105,000 LBS.
LH₂ VOL 4,200 FT³
LO₂ VOL = 4300 FT³
30
12°
198
INTERSTAGE LENGTH
THRUST + COMBINED FORC
AT 3 POINTS ON THE INT
ADAPTER
LENGTH 197.5 FT.

2


2 P & W ENGINES
THRUST 300,000 LBS. EA
OPTIONAL ENGINE
THRUST 600,000 LBS.
MOUNT GIMBAL RINGS
240 DIA.
(20 FT.)
PAYLOAD
LH2
LO2
12°
RINGERS
G. MT.
ROLL CONTROL REQUIRED
WHEN A SINGLE ENGINE
IS USED
270
INTERSTAGE LENGTH
STA
0
300,000 LB. THRUST
P & W ENGINE
TANK LENGTH 813 (67.7 FT)
TENSION CONNECTORS AT
ENGINE MOUNT - TYPICAL
BETWEEN 3 BOOSTERS
240 DIA
(20 FT.)
LH2 VOL = 14,200 FT³
LO2 VOL = 4200 FT³
TENSION CONNECTOR
THRUST & COMBINED FORCES REACTED
AT 3 POINTS ON THE INTERSTAGE
ADAPTER
LAUNCH SUPPORT PYLONS AT 6 POINTS
ASSUMED TO BE UNIFORMLY LOADED
STAGE I
PROPELLANT WT. 330,000 x 3 = 990,000 LBS.
REAR
OF 3 MODULE
CONFIGURATION
4 - A

3

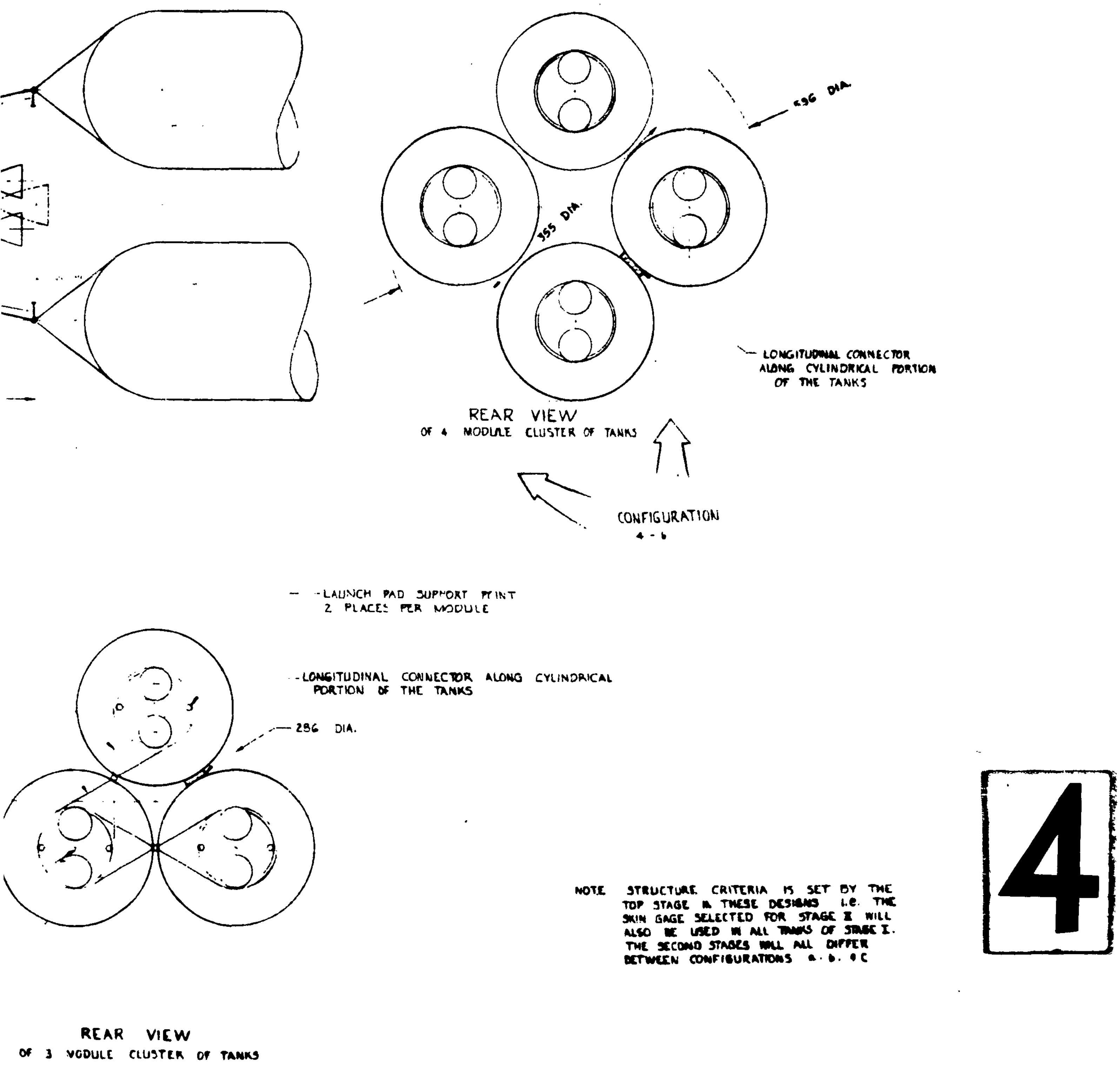
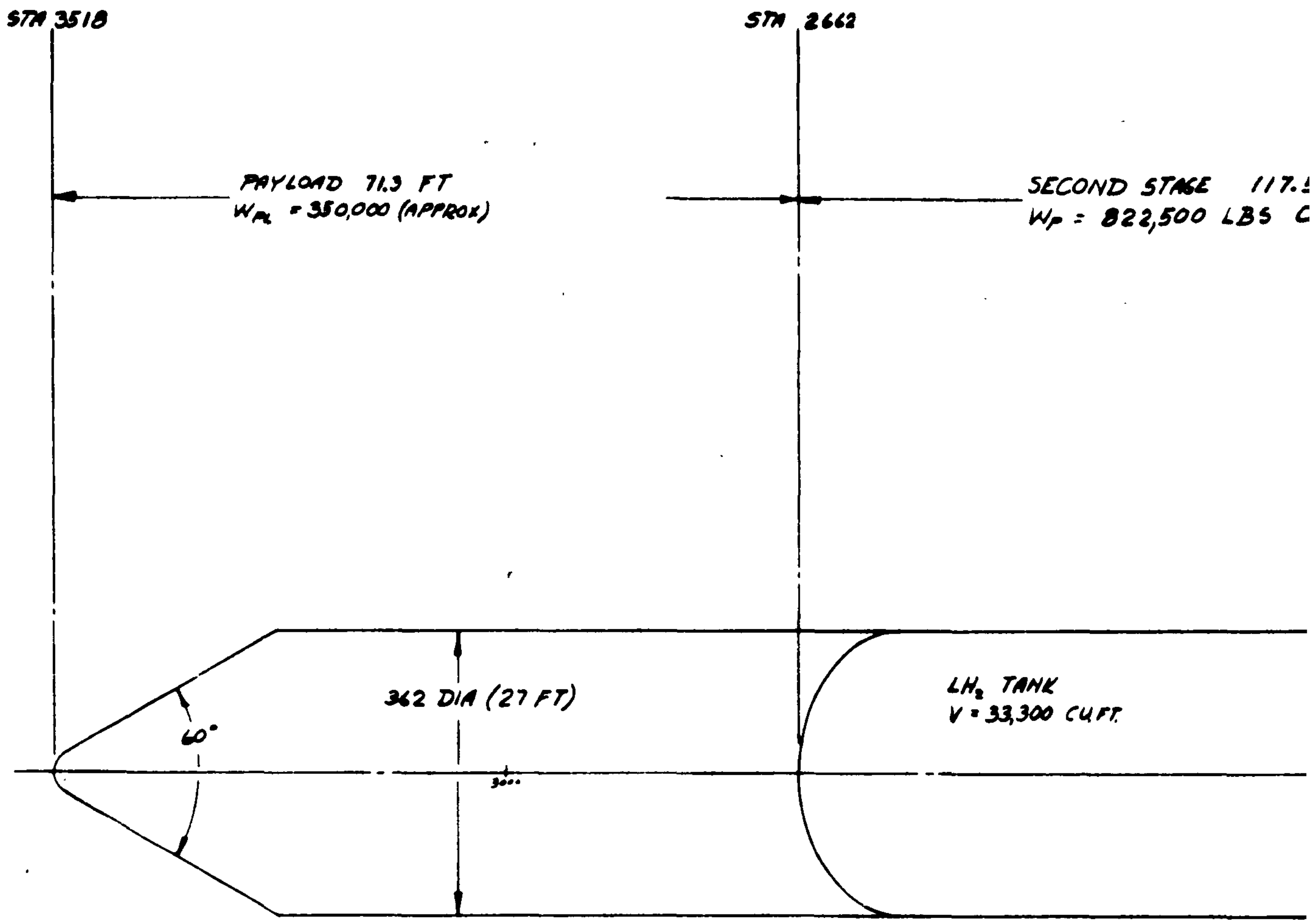
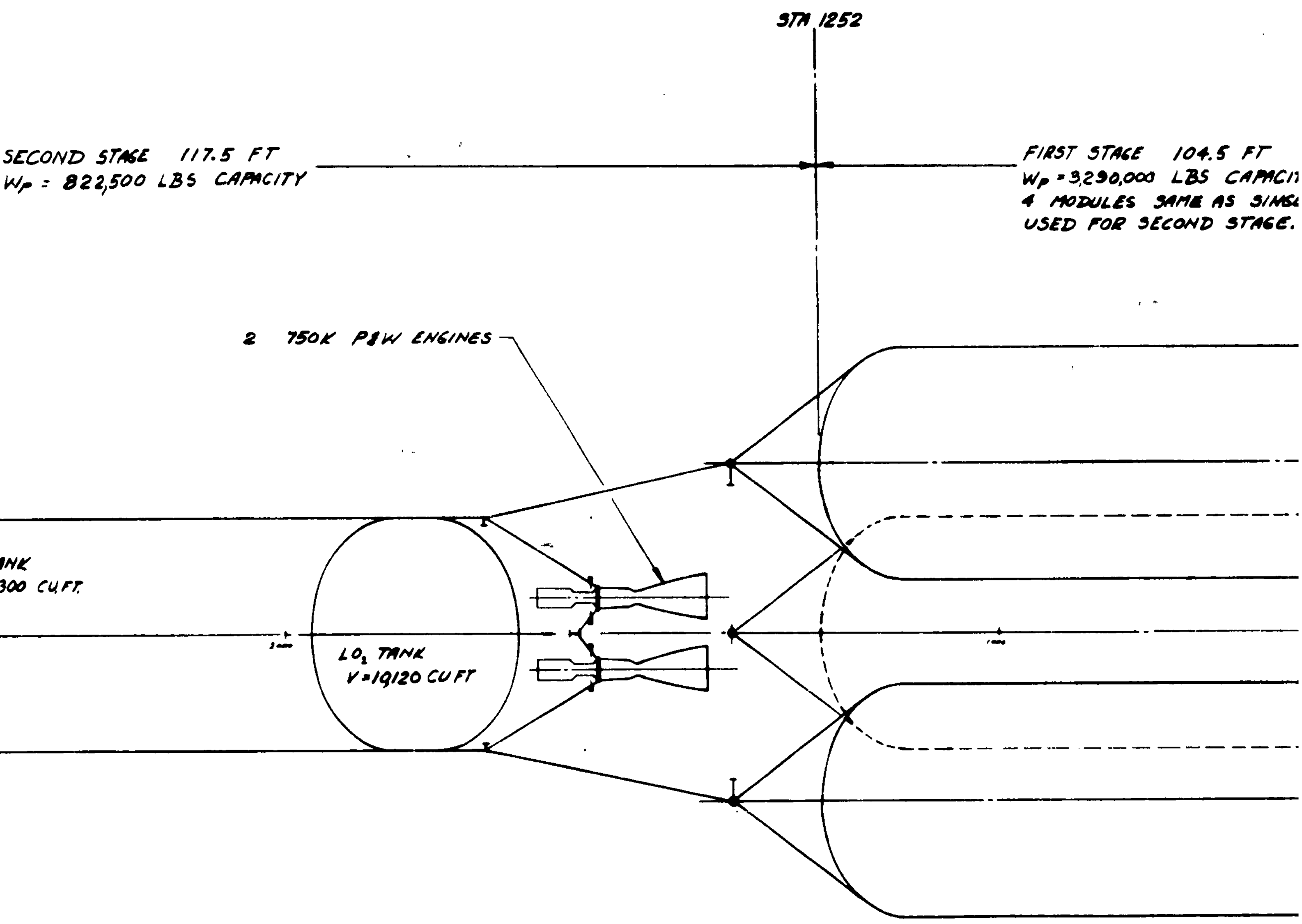


Figure A-7. 600K Pratt & Whitney Engines - Cost Optimized Booster Systems Study Modular Design

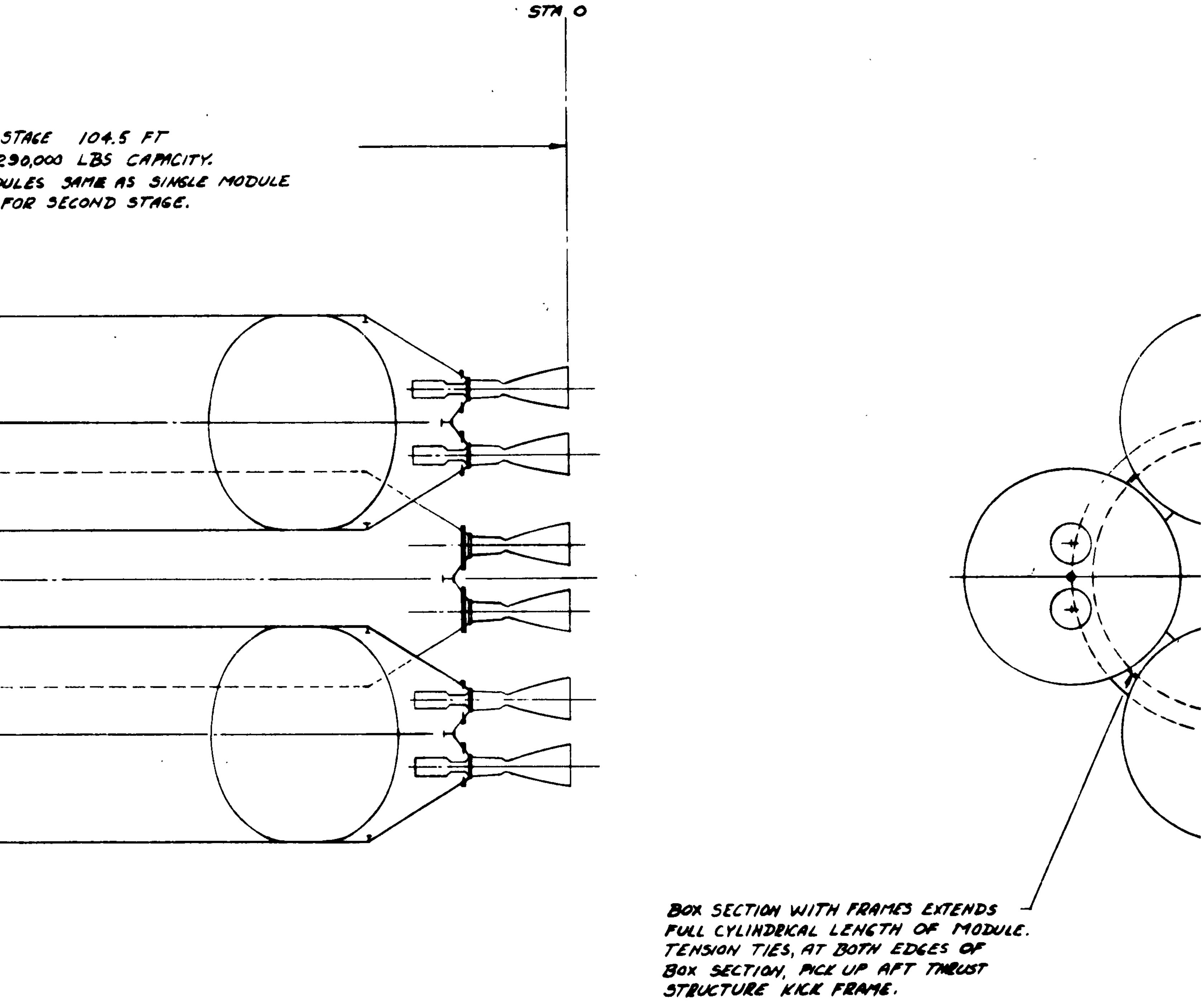
[redacted]


STA 3518
STA 2662
PAYLOAD 71.3 FT
W_PL = 350,000 (APPROX)
SECOND STAGE 117.
Wp = 822,500 LBS
362 DIA (27 FT)
60°
LH2 TANK
V = 33,300 CU FT

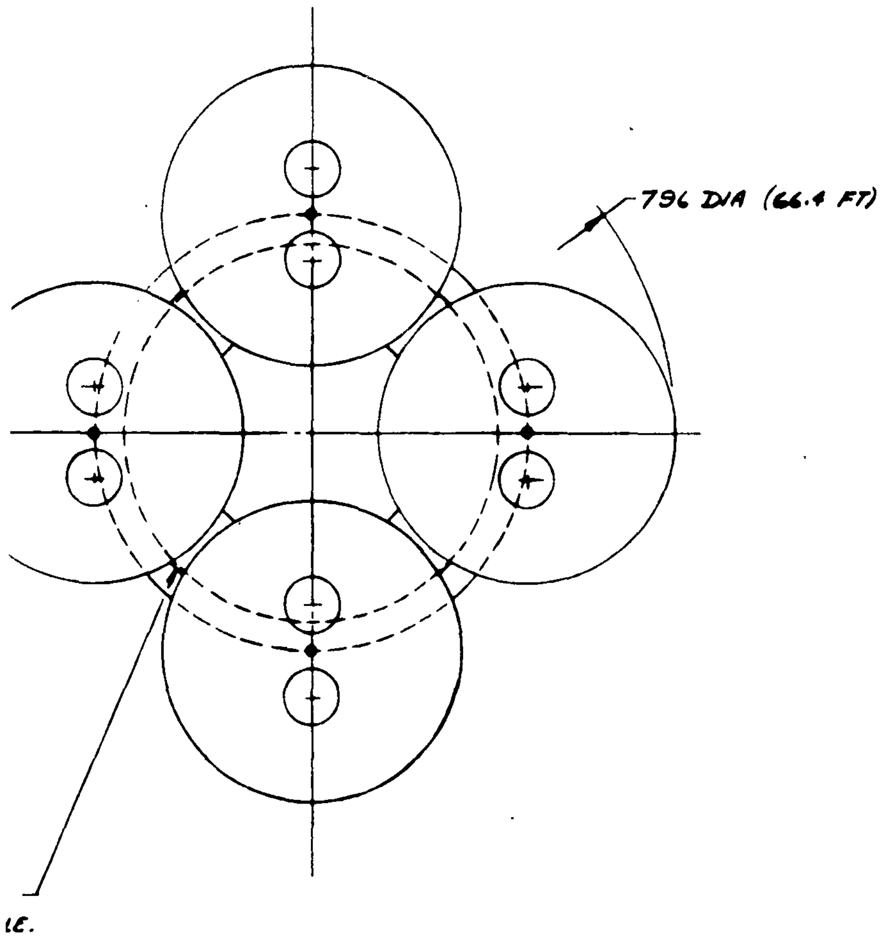
1

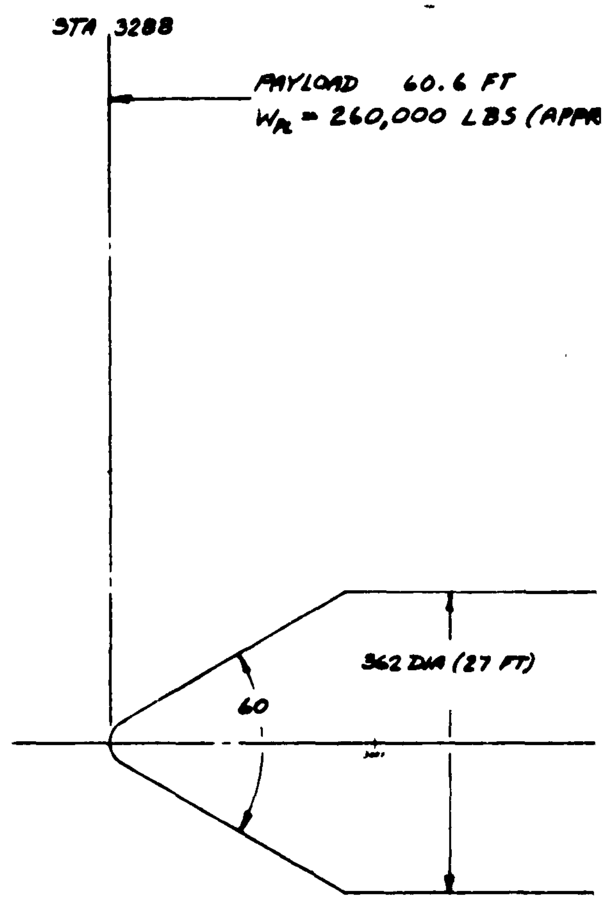

STA 1252
SECOND STAGE 117.5 FT
W_P = 822,500 LBS CAPACITY
FIRST STAGE 104.5 FT
W_P = 3,290,000 LBS CAPACI
4 MODULES SAME AS SINGL
USED FOR SECOND STAGE.
2 750K P&W ENGINES
NK
300 CU FT.
LO_2 TANK
V=10,120 CU FT

2

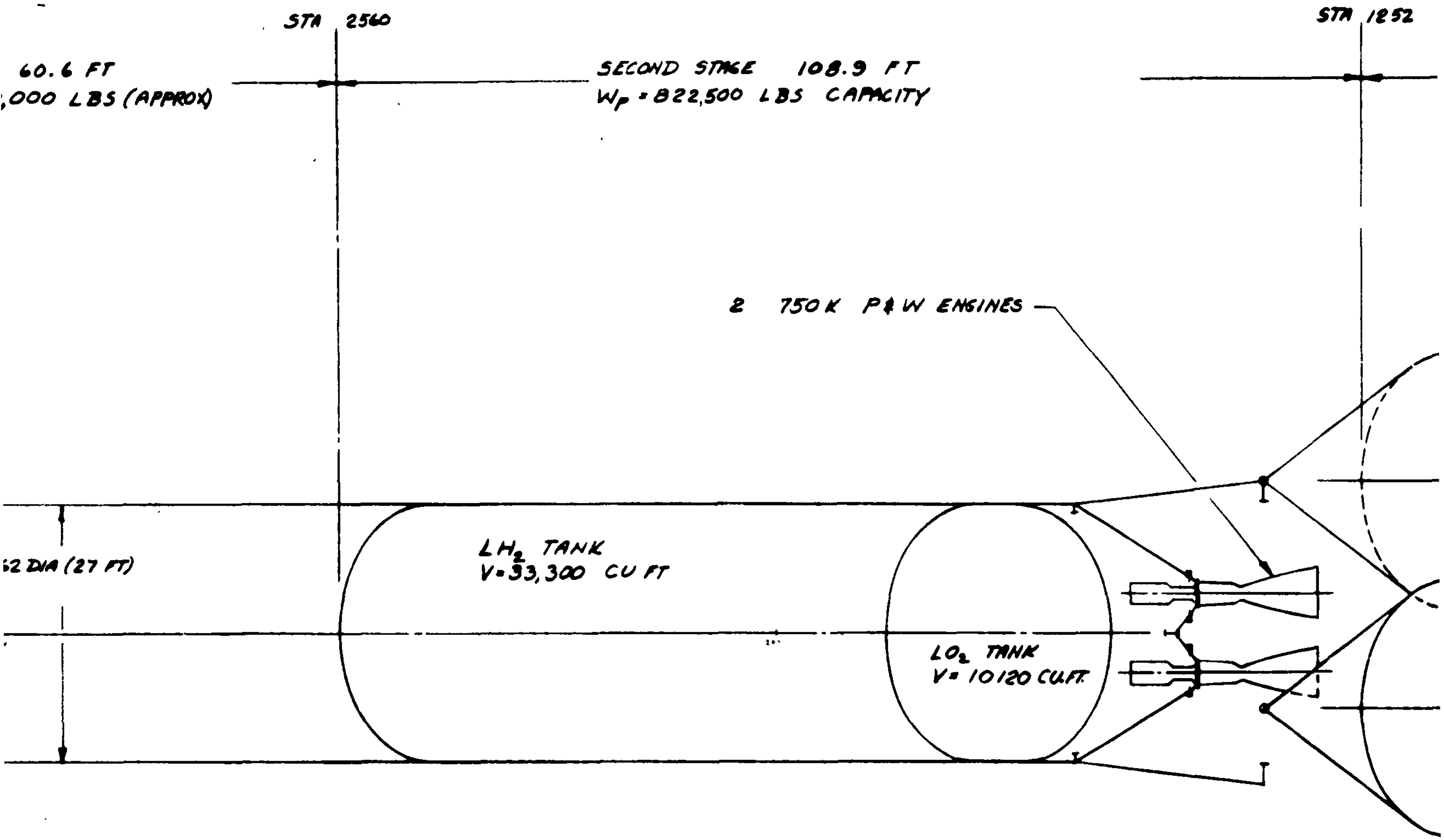

STA 0
STAGE 104.5 FT
290,000 LBS CAPACITY.
ULES SAME AS SINGLE MODULE
FOR SECOND STAGE.
BOX SECTION WITH FRAMES EXTENDS
FULL CYLINDRICAL LENGTH OF MODULE.
TENSION TIES, AT BOTH EDGES OF
BOX SECTION, PICK UP AFT THRUST
STRUCTURE KICK FRAME.

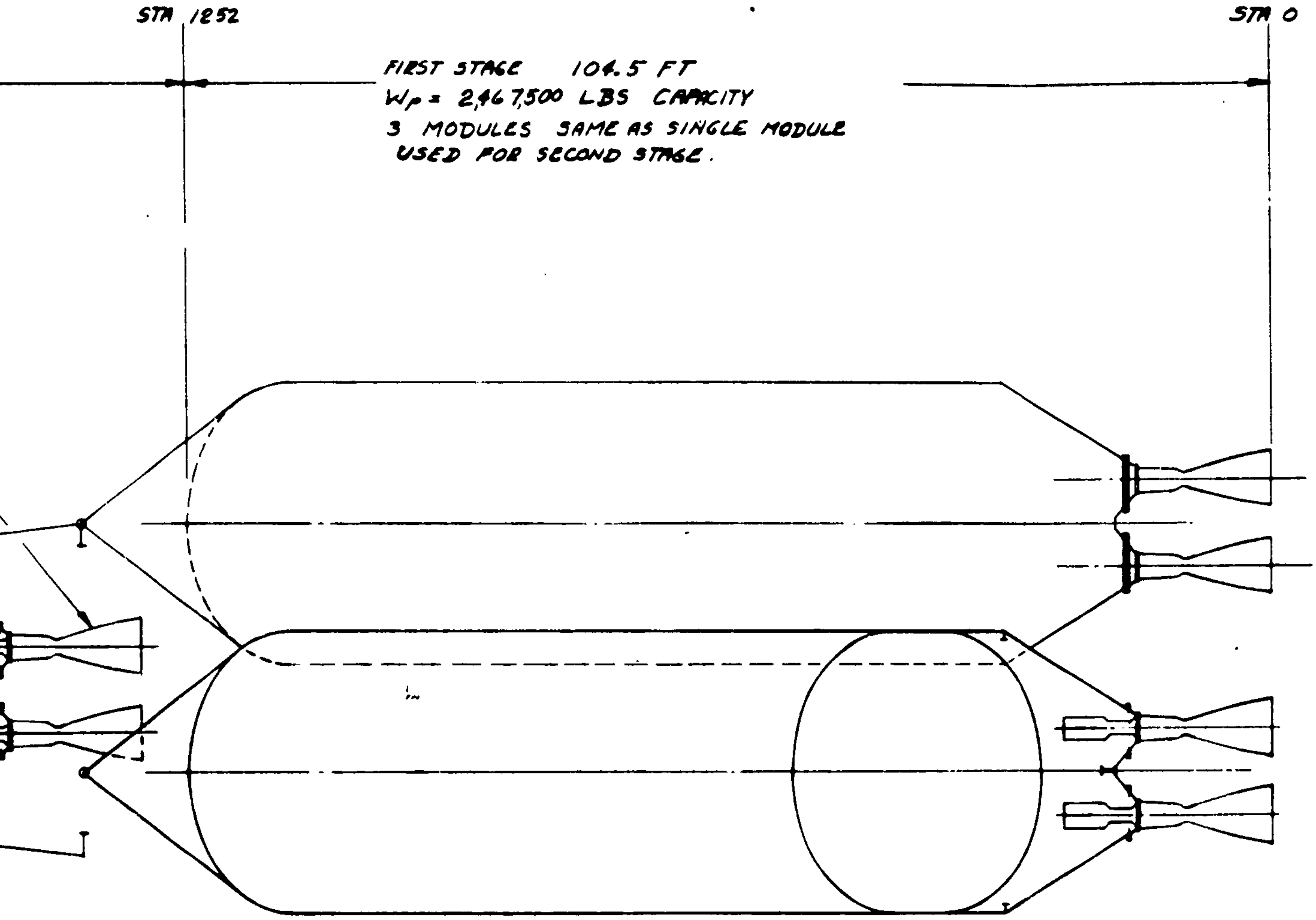
3


796 DIA (66.4 FT)
LE.

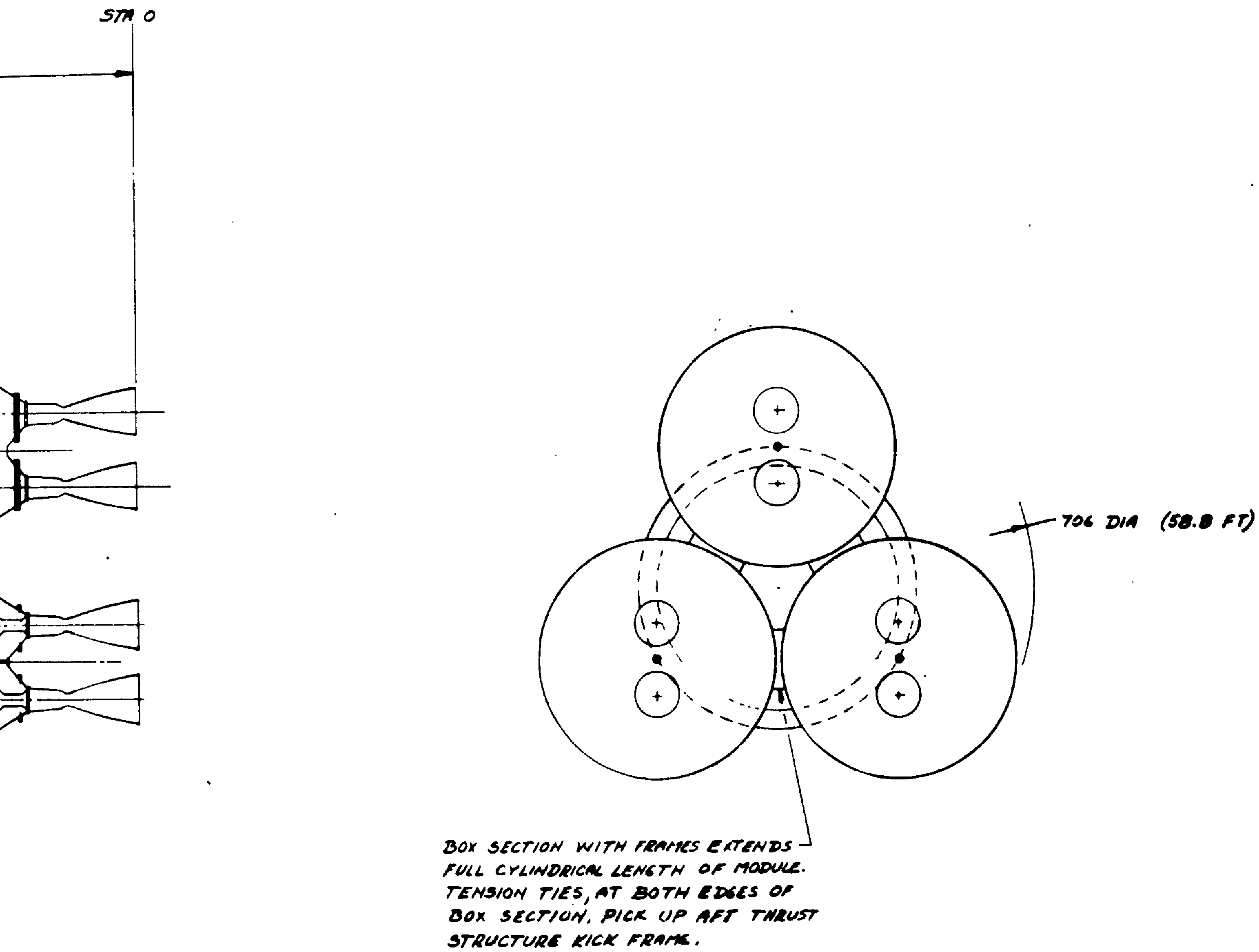

STA 3288
PAYLOAD 60.6 FT
W_{PL} = 260,000 LBS (APPR
362 DIA (27 FT)
60

4


STA 2560
STA 1252
60.6 FT
,000 LBS (APPROX)
SECOND STAGE 108.9 FT
W_p = 822,500 LBS CAPACITY
2 750K P&W ENGINES
2 DIA (27 FT)
LH_2 TANK
V=33,300 CU FT
LO_2 TANK
V= 10120 CU.FT.


STA 1252
STA 0
FIRST STAGE 104.5 FT
W_p = 2,467,500 LBS CAPACITY
3 MODULES SAME AS SINGLE MODULE
USED FOR SECOND STAGE.

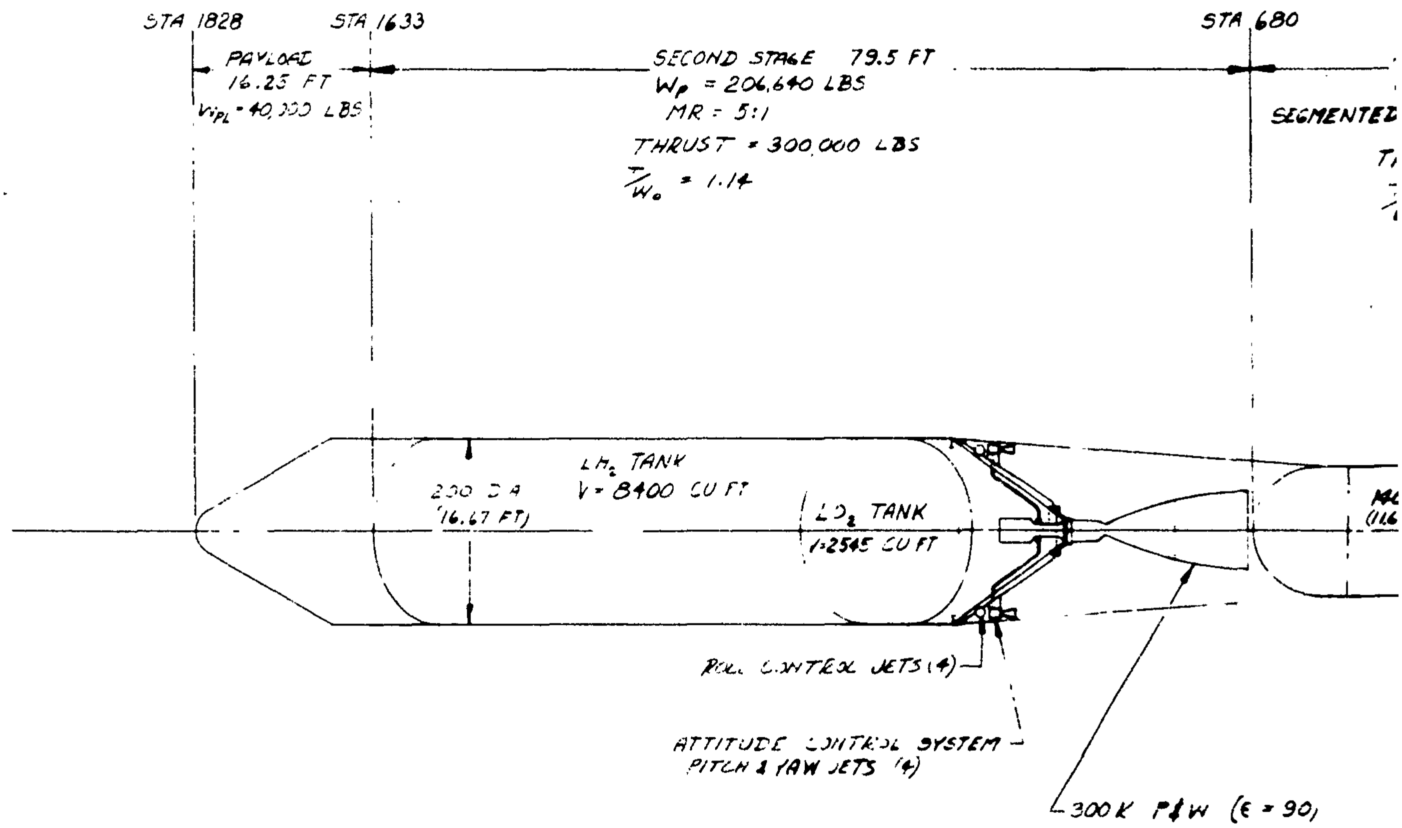
6



7

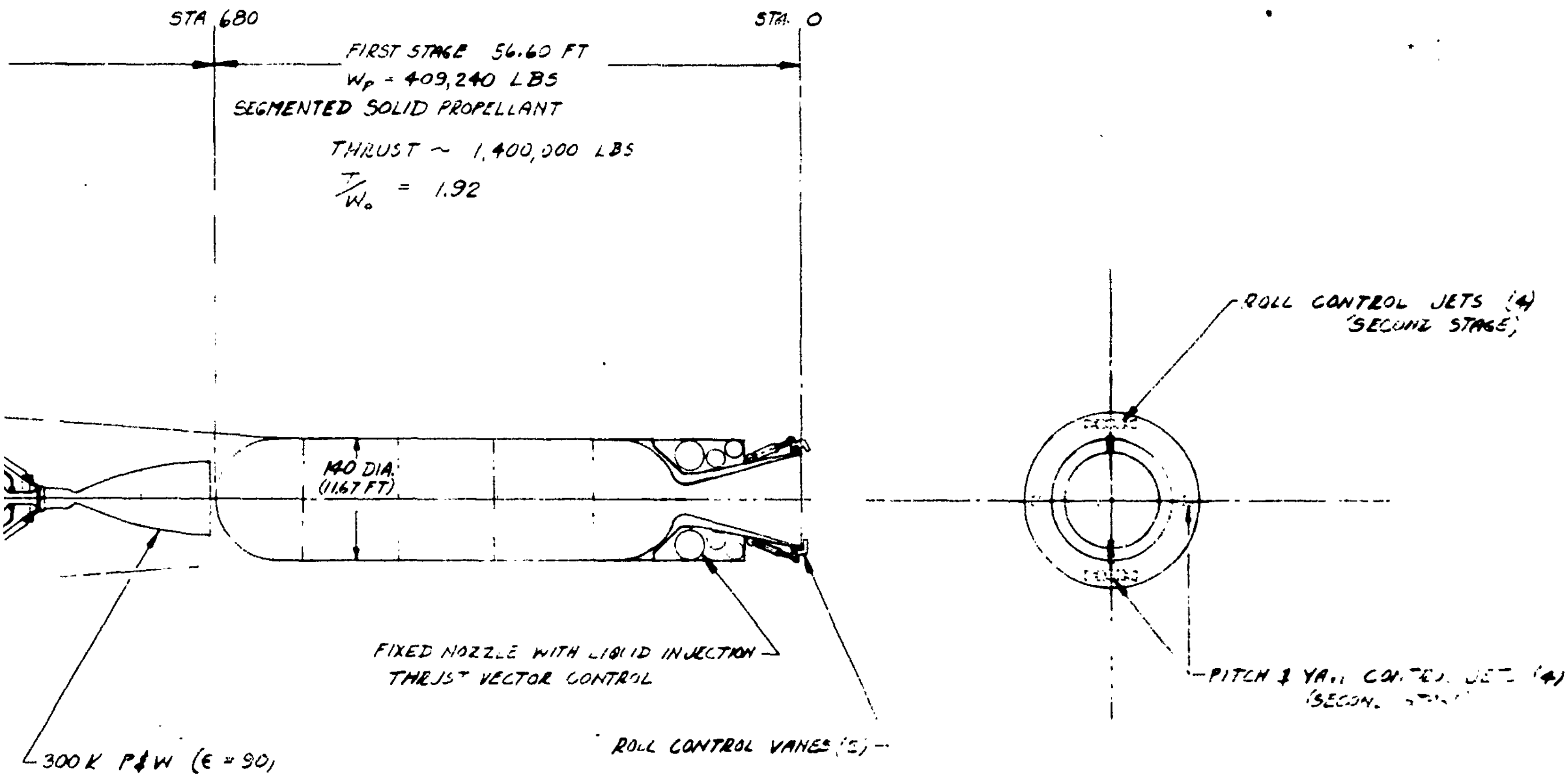
Figure A-8. Modular Design - Three 1500K Modules and Four 1500K Modules with 1500K Second Stages

[redacted]


STA 1828
STA 1633
STA 680
PAYLOAD
16.25 FT
SECOND STAGE 79.5 FT
W_p = 206,640 LBS
MR = 5:1
THRUST = 300,000 LBS
T/W_o = 1.14
SEGMENTED
LH_2 TANK
V = 8400 CU FT
(16.67 FT)
LO_2 TANK
V = 2545 CU FT
CONTROL JETS (4)
ATTITUDE CONTROL SYSTEM
PITCH & YAW JETS (4)
300K P&W

SID 61-341

1

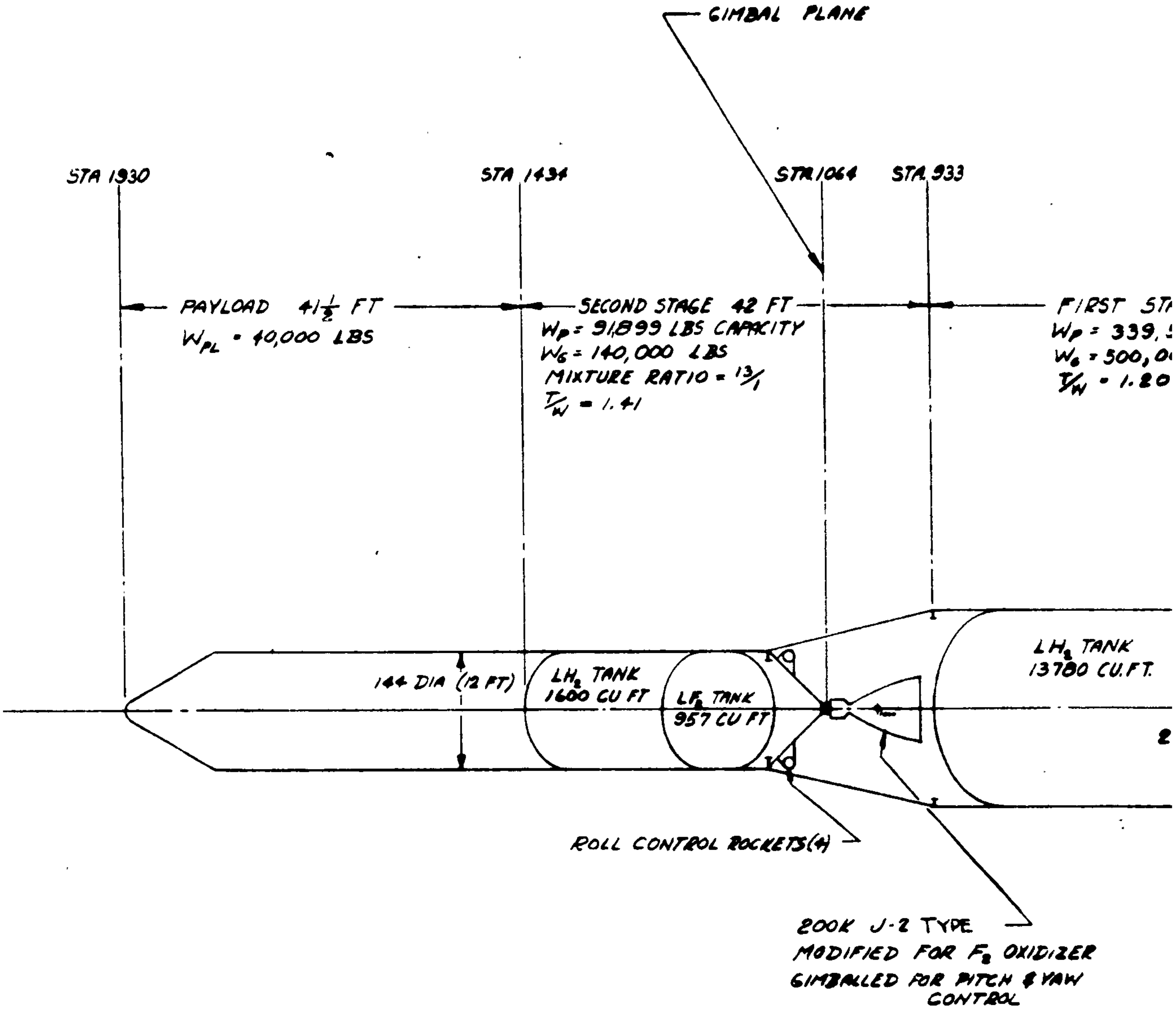


2

Figure A-9. 1.4M Solid First Stage - Pratt & Whitney 300K Liquid Oxygen Liquid Hydrogen Second Stage

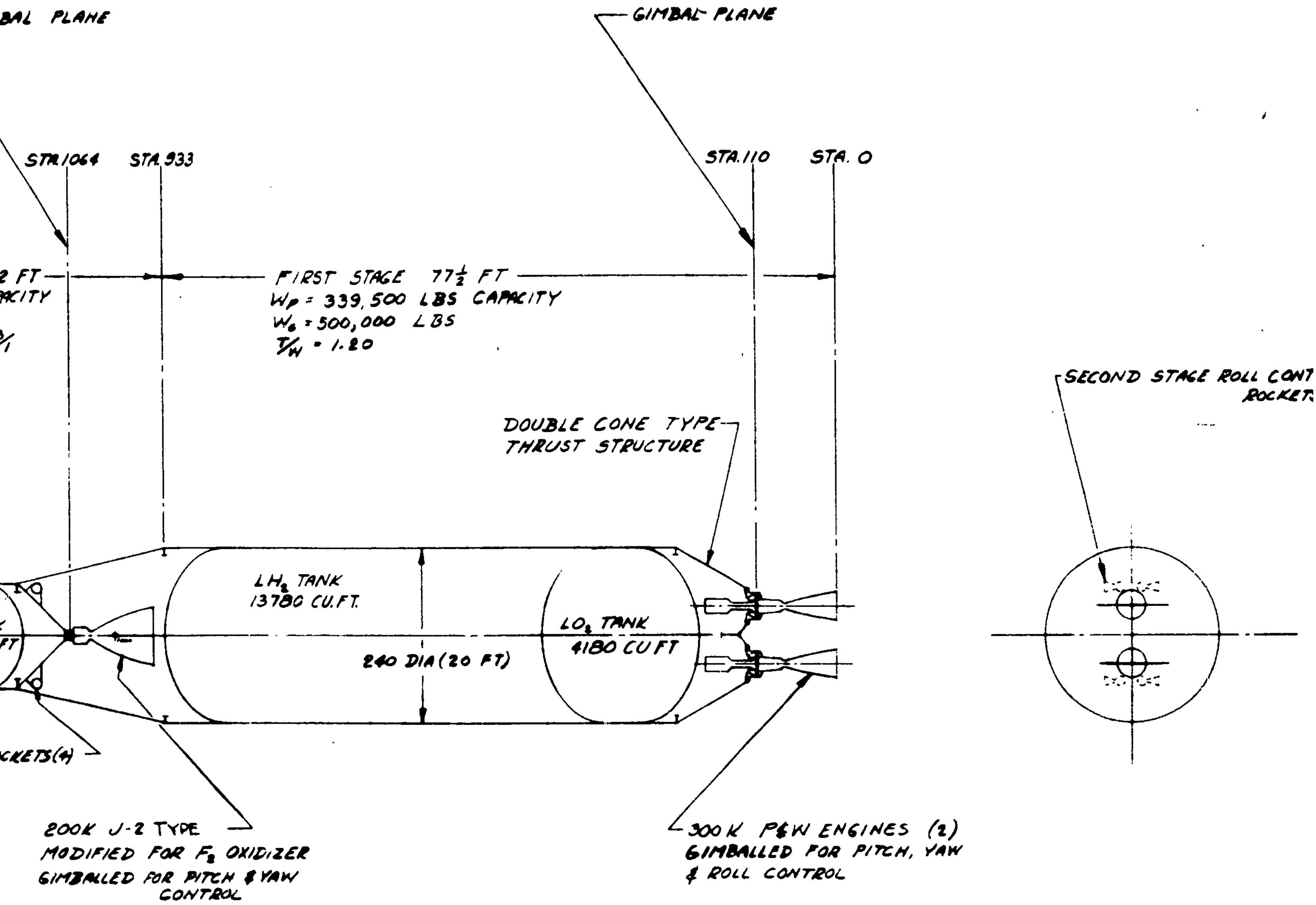
[redacted]

[redacted]



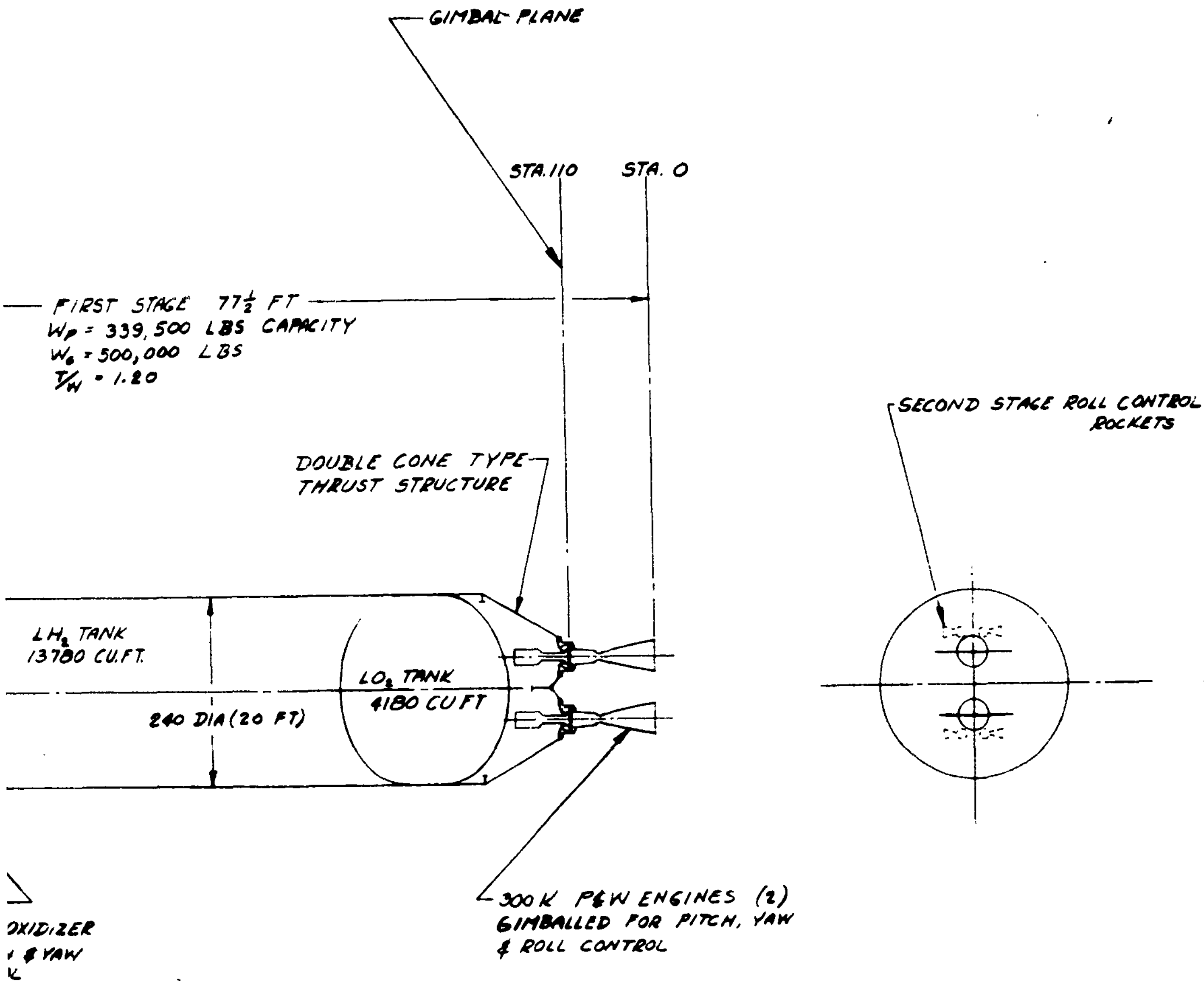
1

SID 61-341



2

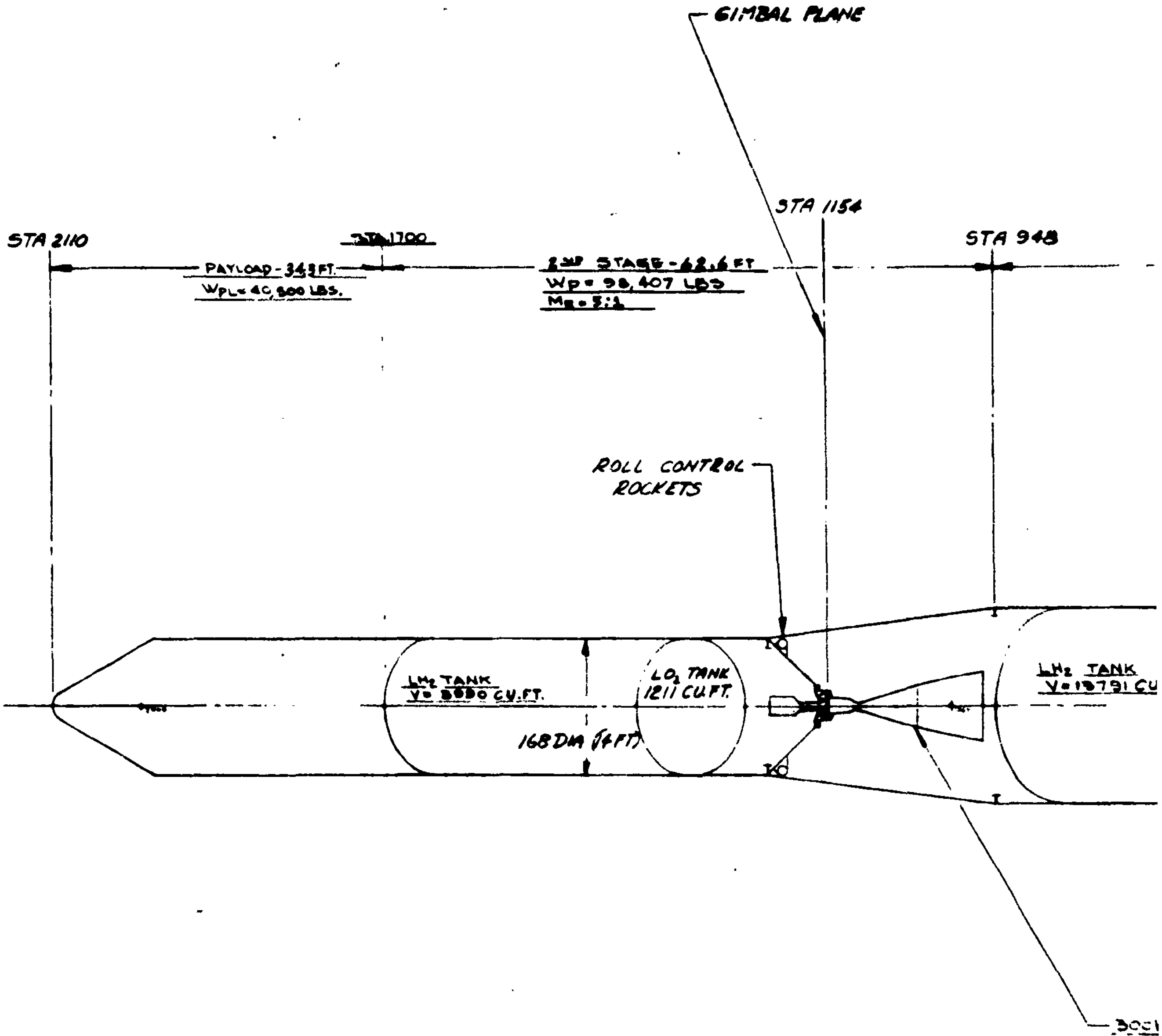
Figure A-10. 600K Liquid Oxygen/Liquid Hydrogen Pratt & Whitney First Stage and 200K Liquid Flouorine/Liquid Hydrogen Second Stag



3

Figure A-10. 600K Liquid Oxygen/Liquid Hydrogen Pratt & Whitney First Stage and 200K Liquid Flouorine/Liquid Hydrogen Second Stage

[redacted]


GIMBAL PLANE
STA 2110
STA 1700
STA 1154
STA 948
PAYLOAD - 34.2 FT.
W_{PL} = 40,800 LBS.
2ND STAGE - 62.6 FT
Wp = 98,407 LBS
ROLL CONTROL ROCKETS
LH_2 TANK
V = 3030 CU.FT.
LO_2 TANK
1211 CU.FT.
168 DIA (14 FT)
LH_2 TANK
V = 13791 CU

1

SID 61-341

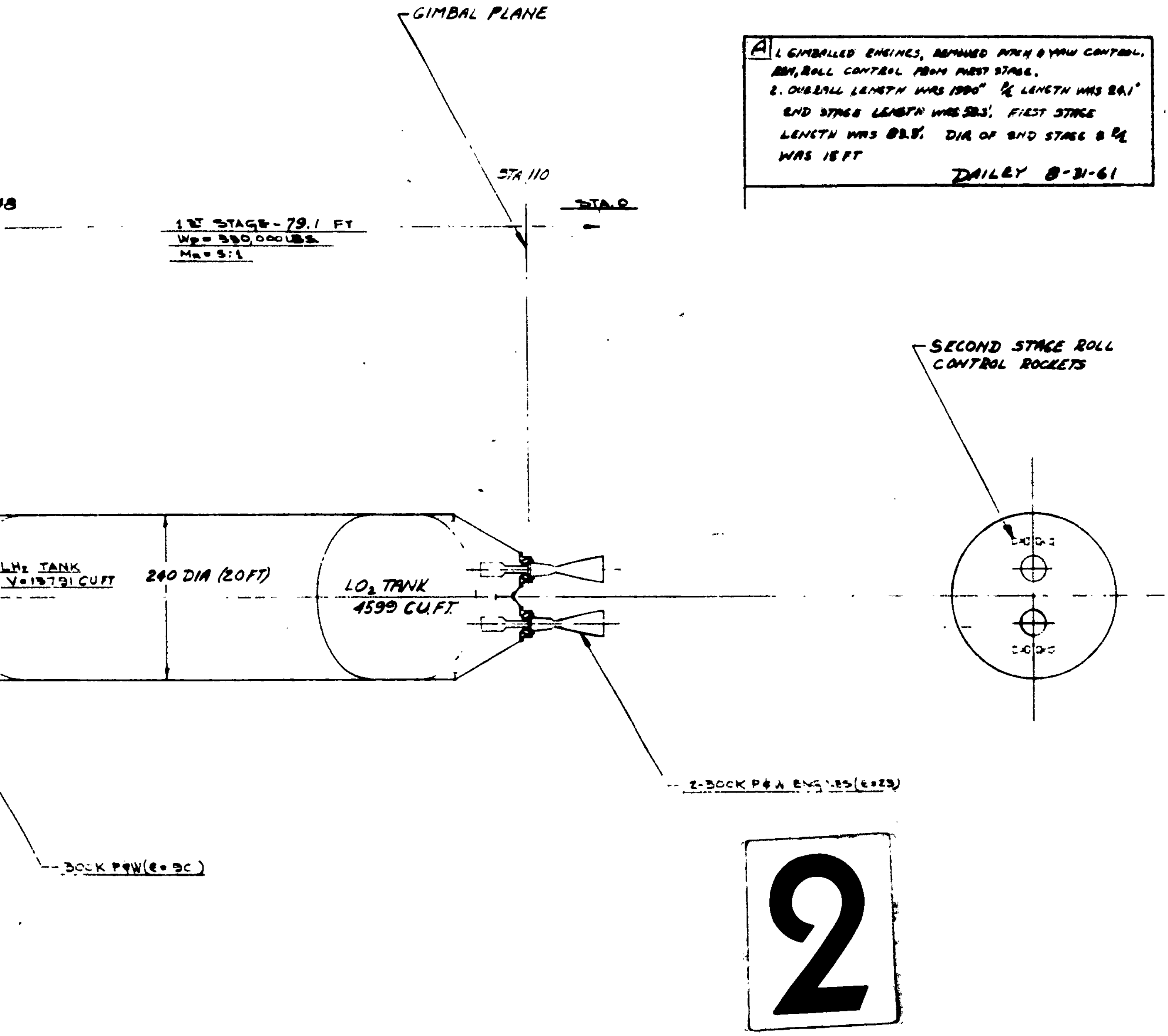


Figure A-11. 600K Pratt & Whitney First Stage - 300K Pratt & Whitney Second Stage

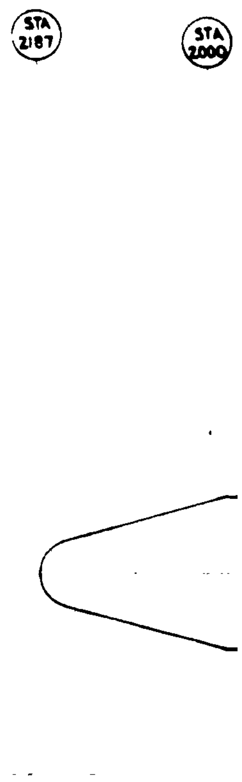
[redacted]

[redacted]

DOUGLAS 3 TANK DESIGN
PARALLEL STAGING (FROM DOUGLAS REPORT)

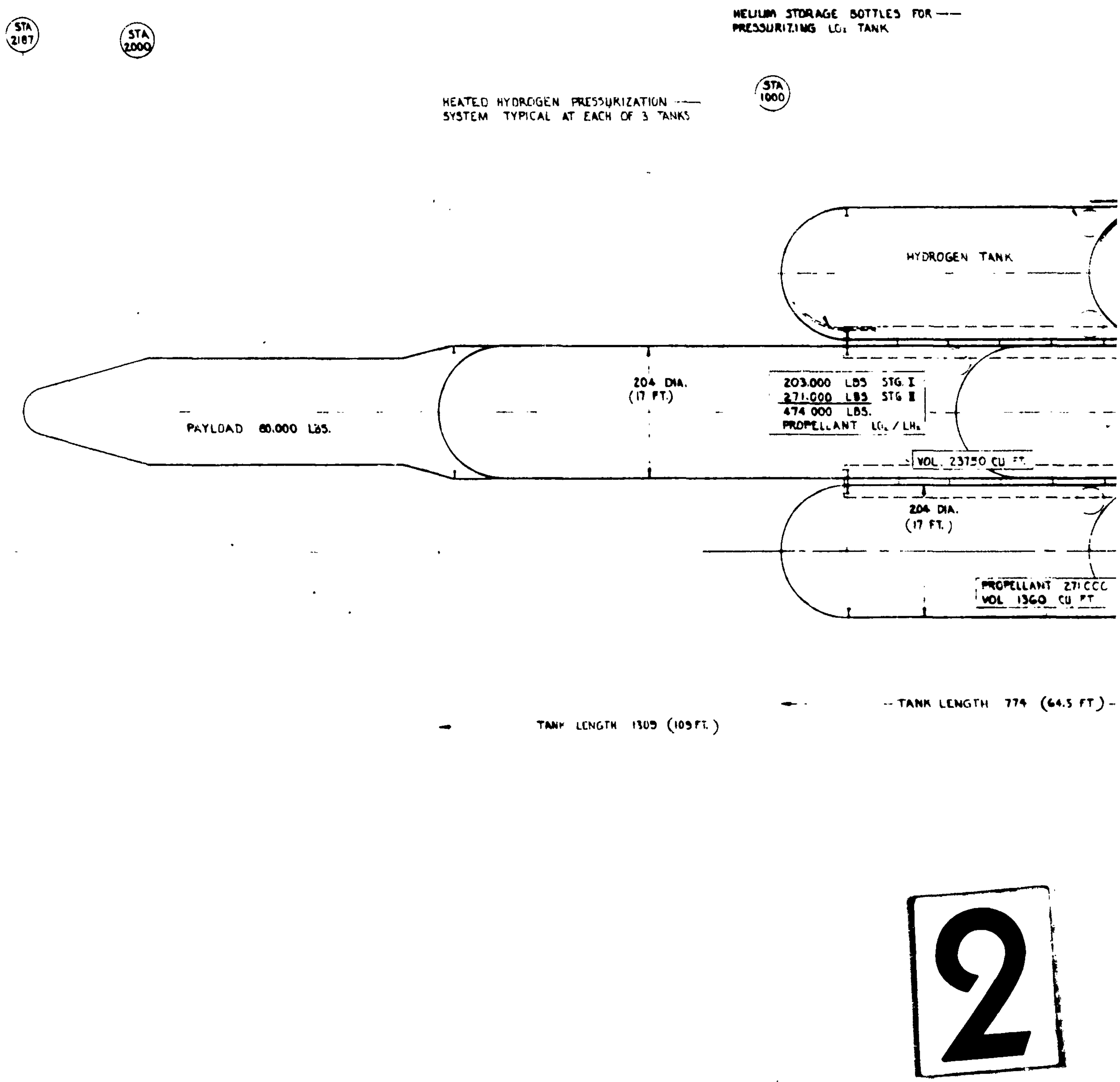
PAYLOAD		70,000 LBS
PROPELLANT	— LH_2	385,245
	— LO_2	161,000
AMPR WEIGHT		45,855

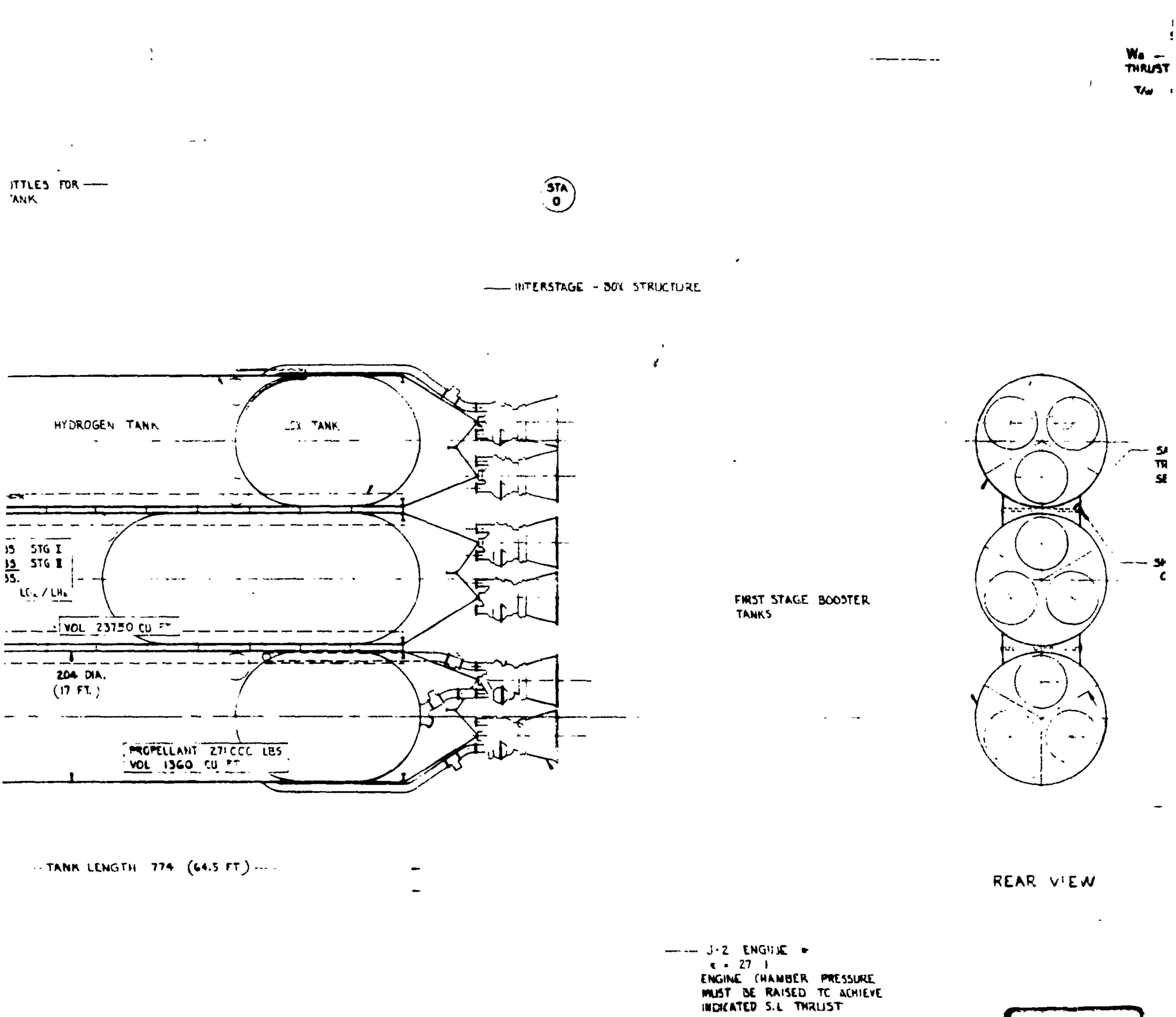
1.5 M # THRUST PLUG ENGINE



SID 61-341

1


STA 2187
STA 2000
HELIUM STORAGE BOTTLES FOR PRESSURIZING LO2 TANK
STA 1000
HEATED HYDROGEN PRESSURIZATION SYSTEM TYPICAL AT EACH OF 3 TANKS
HYDROGEN TANK
204 DIA. (17 FT.)
203.000 LBS STG. I
271.000 LBS STG II
474 000 LBS.
PROPELLANT LO2 / LH2
PAYLOAD 60.000 LBS.
VOL. 23750 CU FT
204 DIA. (17 FT.)
PROPELLANT 271 CCC
VOL 1360 CU FT
TANK LENGTH 774 (64.5 FT)
TANK LENGTH 1309 (109 FT.)
2


Wo
THRUST
T/w
ITTLES FOR
ANK
STA
0
INTERSTAGE - BOX STRUCTURE
HYDROGEN TANK
LOX TANK
STG I
STG II
LO₂ / LH₂
VOL 23750 CU FT
204 DIA.
(17 FT.)
PROPELLANT 271CCC LBS
VOL 1360 CU FT
TANK LENGTH 774 (64.5 FT)
FIRST STAGE BOOSTER
TANKS
REAR VIEW
J-2 ENGINE
ENGINE CHAMBER PRESSURE
MUST BE RAISED TC ACHIEVE
INDICATED S.L THRUST

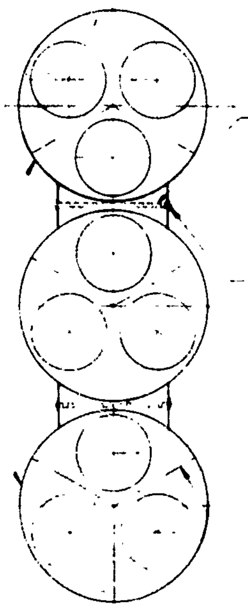
3

WTS. USED FOR LOAD ESTIMATIONS

PAYLOAD	70,000	LBS
PROPELLANT	1,016,000	LBS.
STRUCTURE	113,000	
Wo	1,199,000	LBS.
THRUST	1,500,000	LBS

T/W = 1.25

* THRUST OF EACH ENGINE IS ASSUMED TO BE 166,000 LBS. AT SEA LEVEL ALTHOUGH THE EXPANSION RATIO AS PICTURED IS 27:1



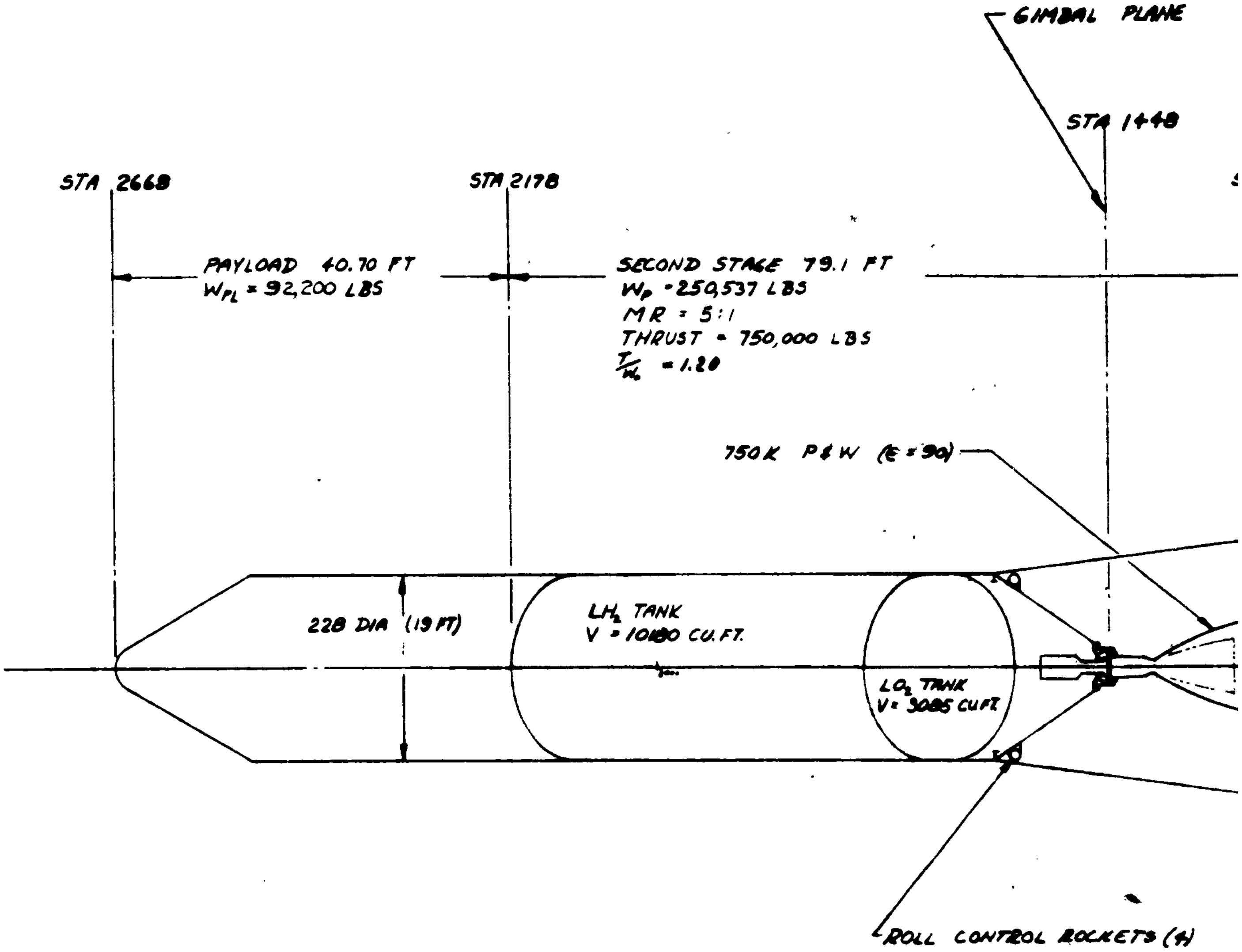
4

Figure A-12. Lateral Staging with J-2 Engines

C.O.B.S. L/
WITH J

[redacted]

[redacted]



SID 61-341

1

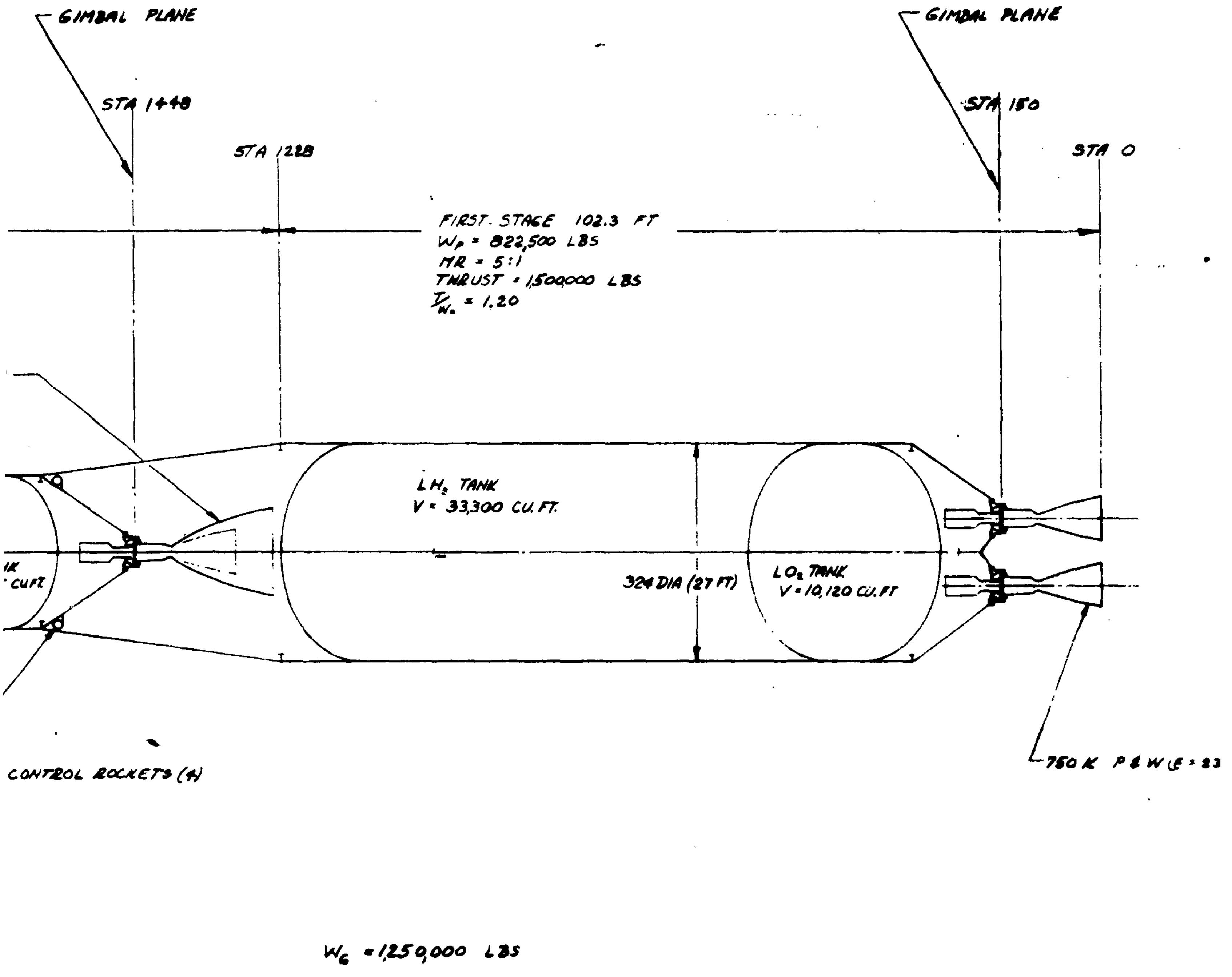

GIMBAL PLANE
STA 1448
STA 1228
GIMBAL PLANE
STA 150
STA 0
FIRST- STAGE 102.3 FT
W_p = 822,500 LBS
MR = 5:1
THRUST = 1,500,000 LBS
T/W_o = 1.20
LH_2 TANK
V = 33,300 CU. FT.
324 DIA (27 FT)
LO_2 TANK
V = 10,120 CU. FT
CONTROL ROCKETS (4)
750 K P & W (ε = 23
W_G = 1,250,000 LBS

Figure 2

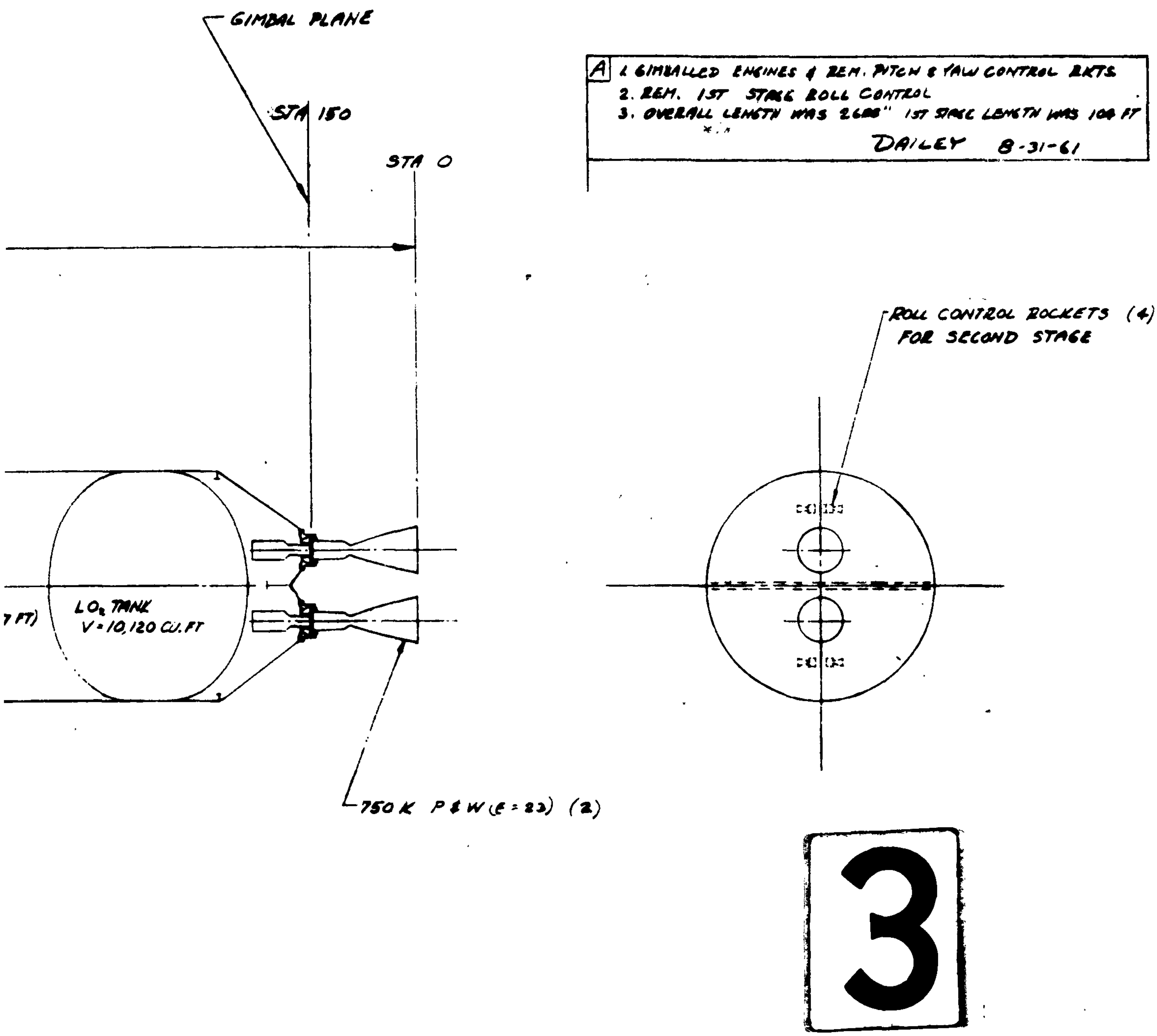
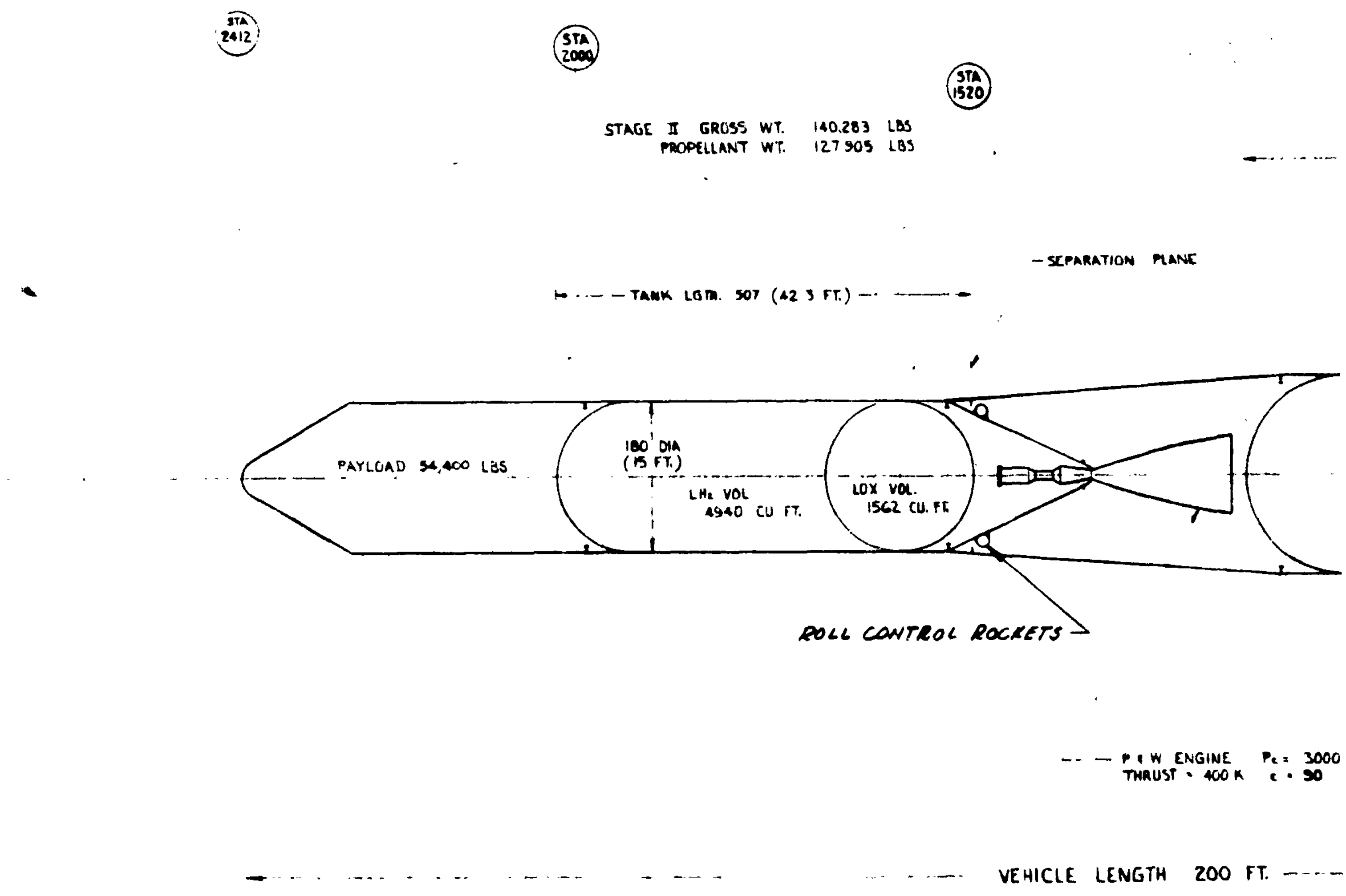


Figure A-13. 1500K Pratt & Whitney First Stage - Pratt & Whitney Second Stage

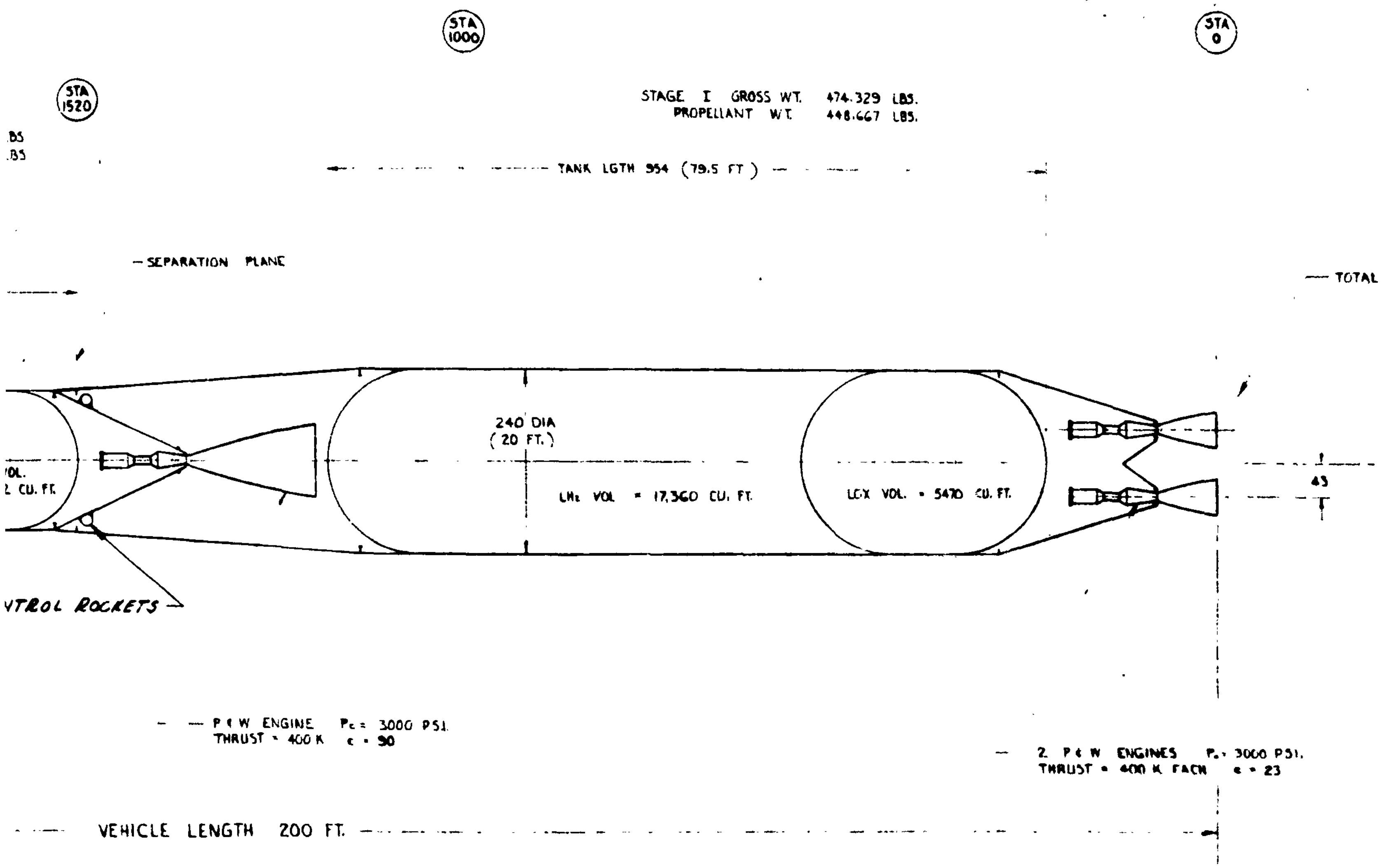
[redacted]

[redacted]



1

SID 61-341



2

Figure A-14. 800K Prat
400K Pratt & Whi

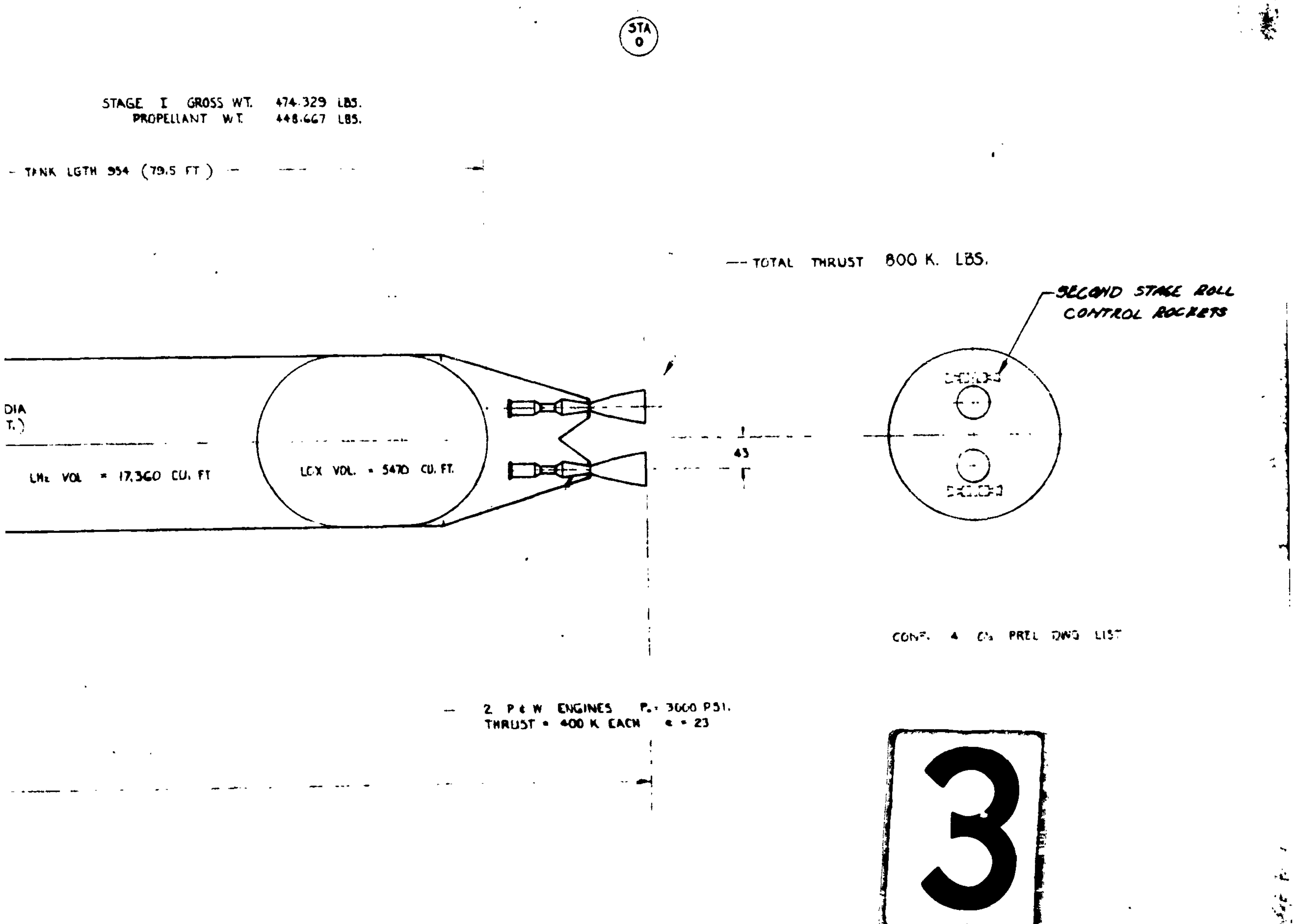
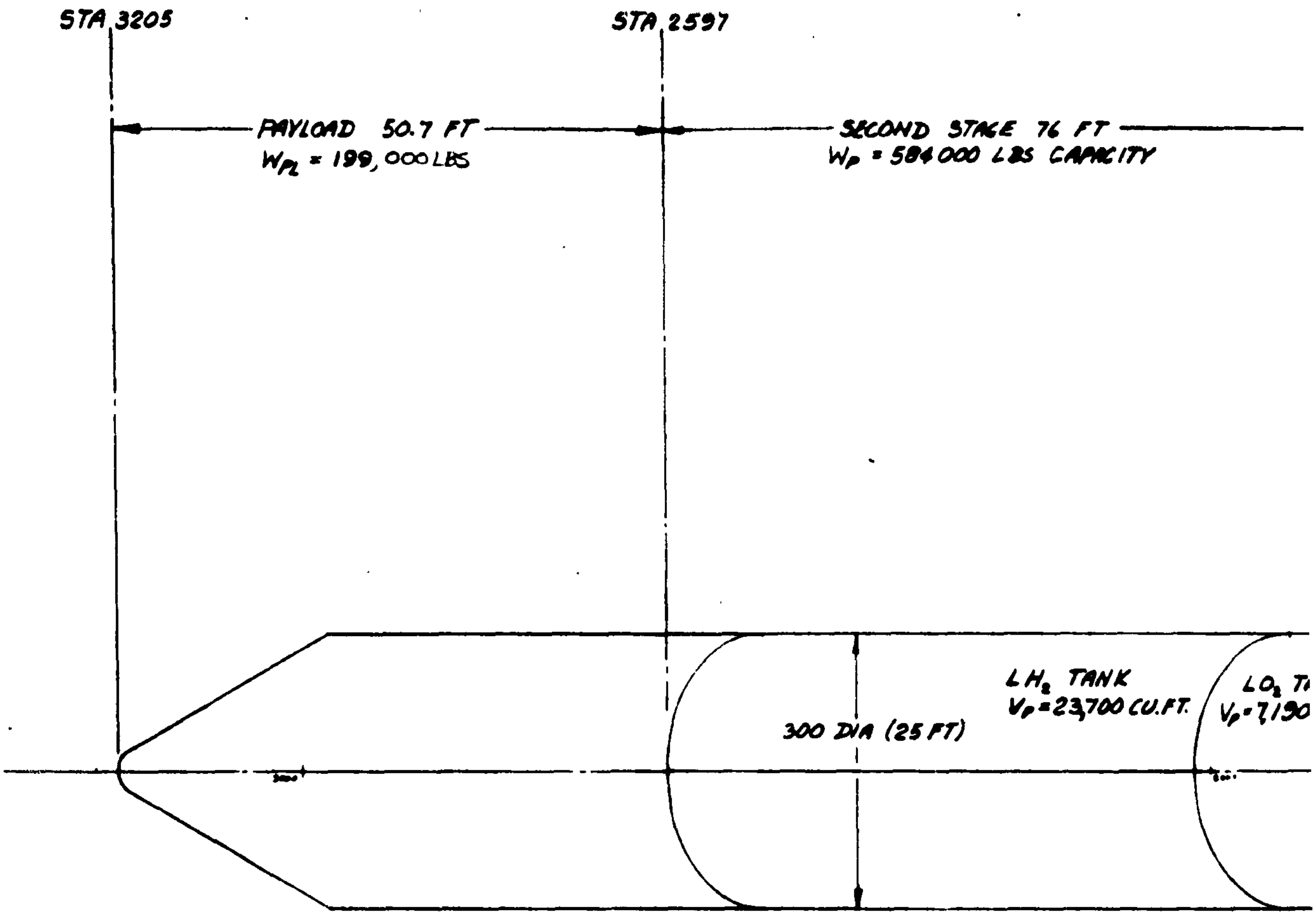


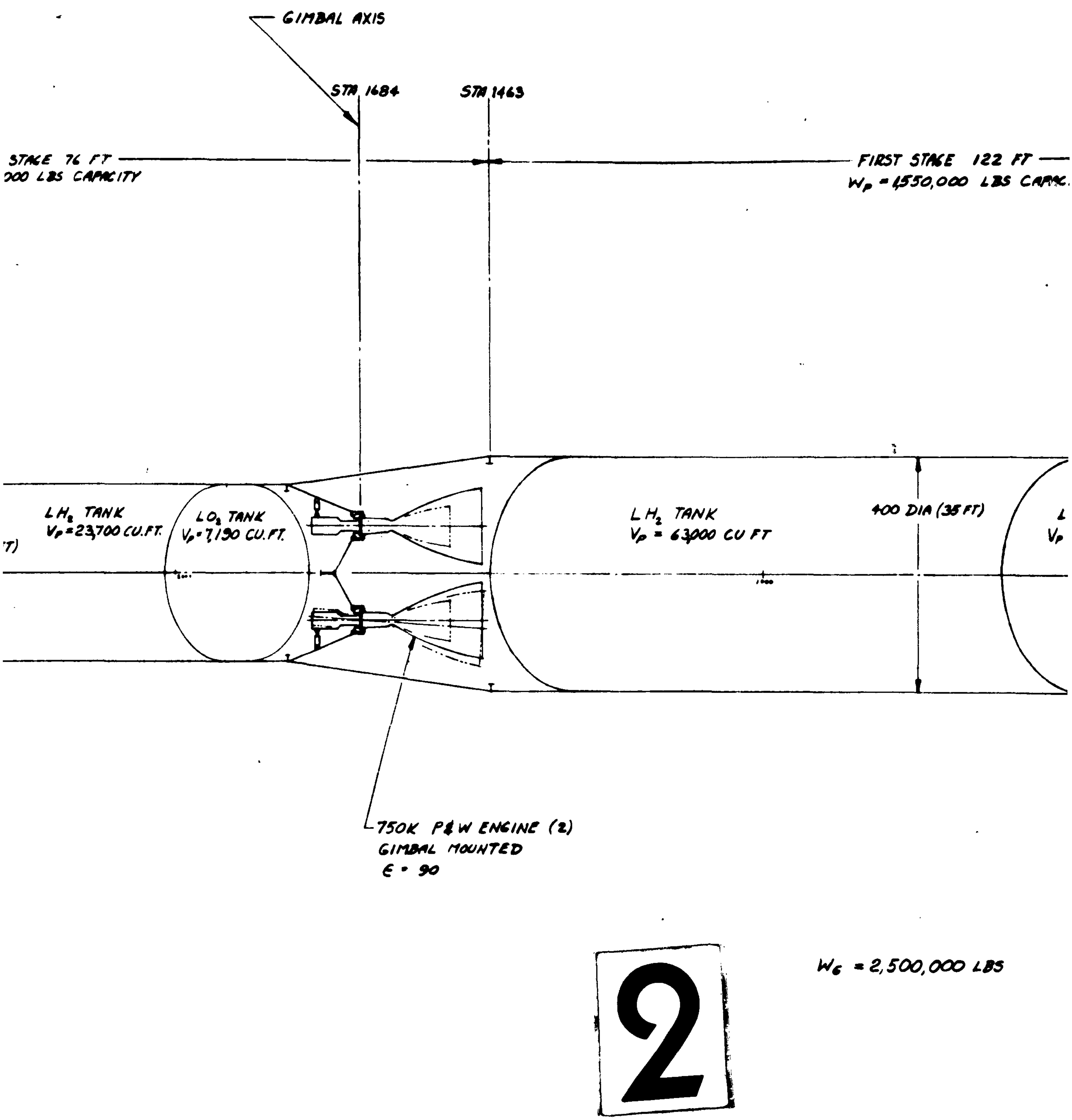
Figure A-14. 800K Pratt & Whitney First Stage - 400K Pratt & Whitney Second Stage

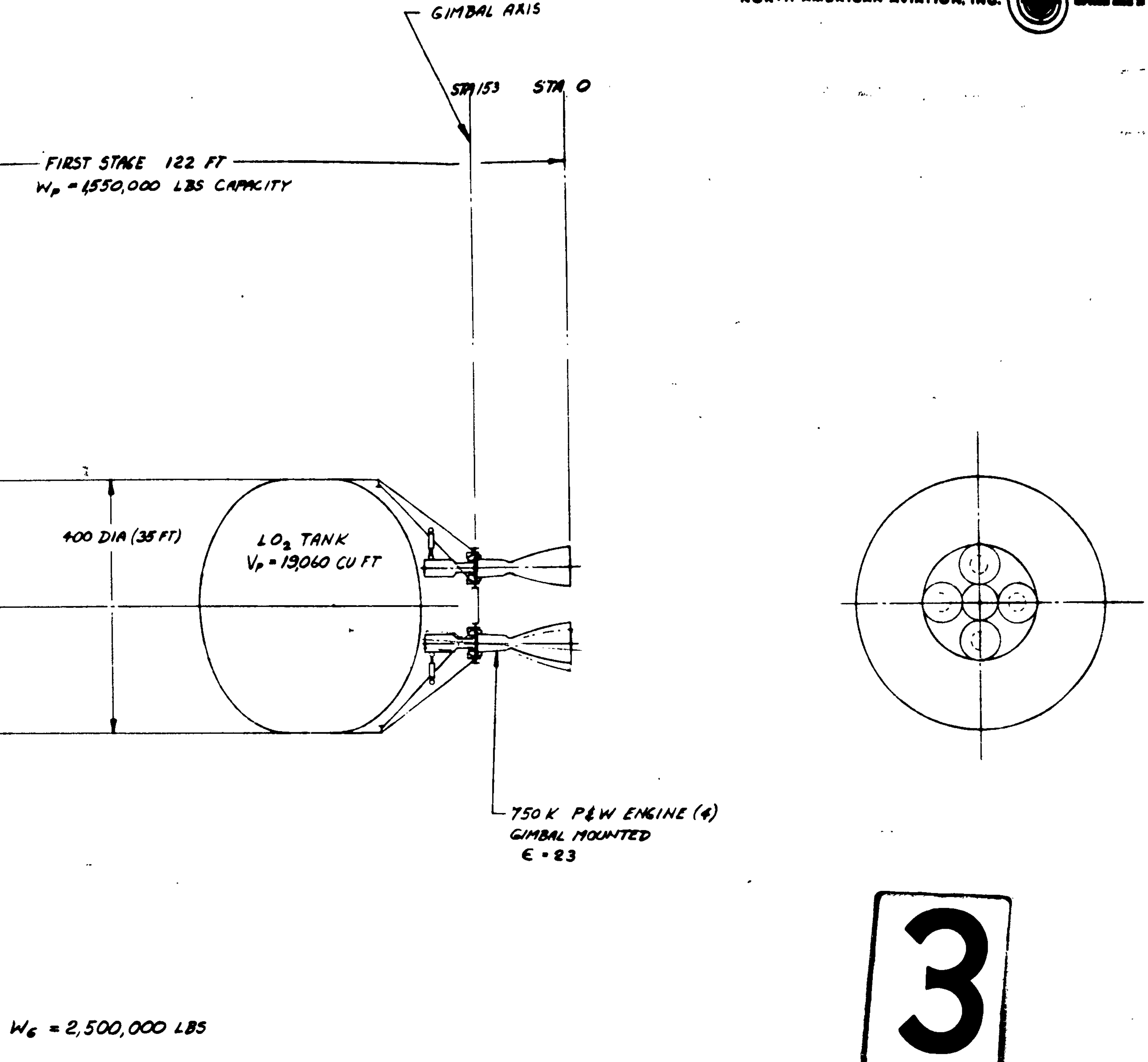
[redacted]

[redacted]


STA 3205
STA 2597
PAYLOAD 50.7 FT
W_{PL} = 199,000 LBS
SECOND STAGE 76 FT
W_P = 584000 LBS CAPACITY
LH_2 TANK
V_P=23,700 CU.FT.
300 DIA (25 FT)

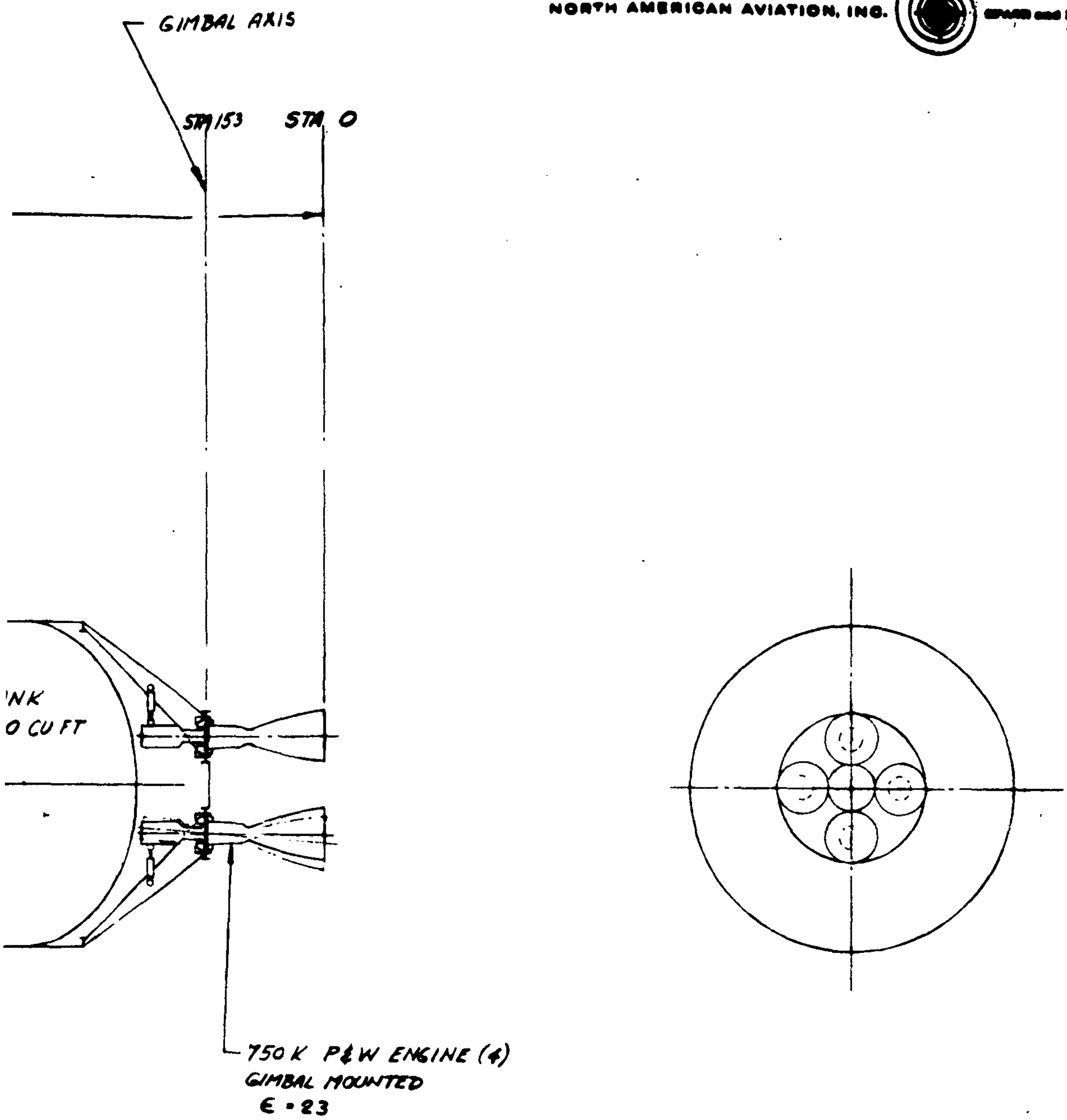
1


GIMBAL AXIS
STA 1684
STA 1463
STAGE 76 FT
000 LBS CAPACITY
FIRST STAGE 122 FT
W_P = 1,550,000 LBS CAPAC.
LH_2 TANK
V_P = 23,700 CU.FT.
LO_2 TANK
V_P = 7,190 CU.FT.
LH_2 TANK
V_P = 63,000 CU FT
400 DIA (35 FT)
750K P&W ENGINE (2)
GIMBAL MOUNTED
ϵ = 90
2
W_G = 2,500,000 LBS



3

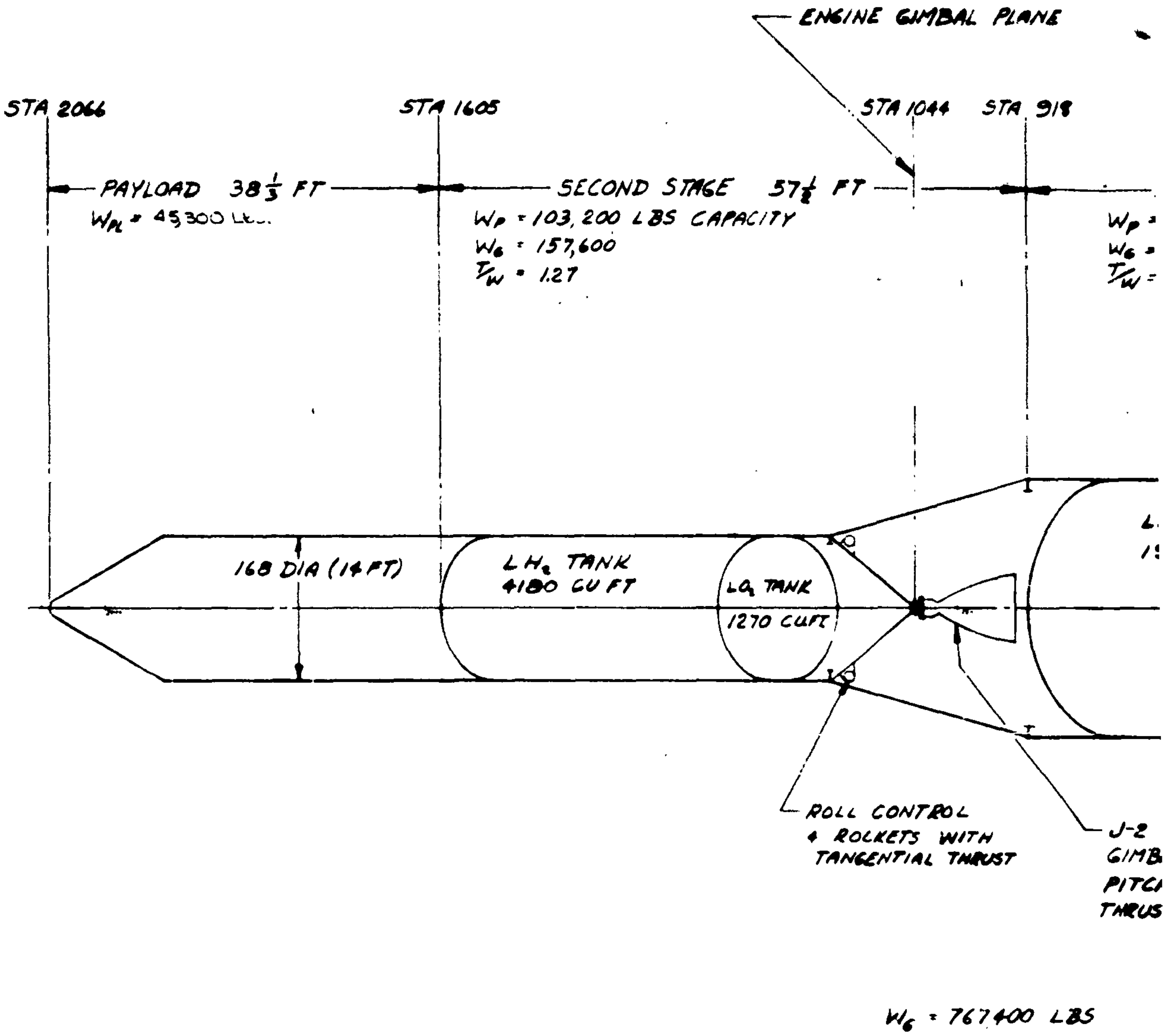
**Figure A-15. 3000K Pratt & Whitney First Stage
1500K Pratt & Whiteny Second Stage**



4

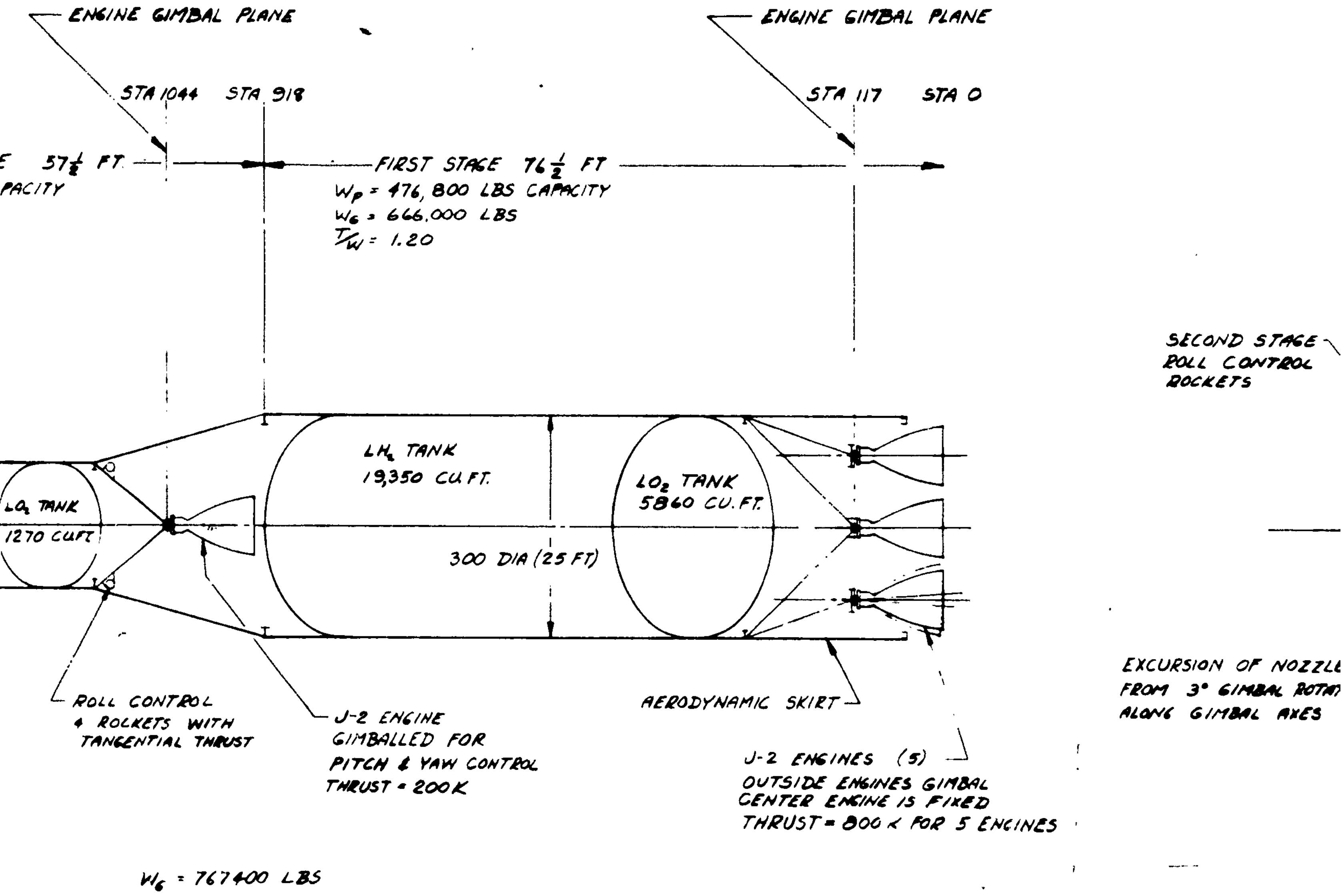
Figure A-15. 3000K Pratt & Whitney First Stage - 1500K Pratt & Whiteny Second Stage

[redacted]


ENGINE GIMBAL PLANE
STA 2066
STA 1605
STA 1044
STA 918
PAYLOAD 38 1/3 FT
W_{PL} = 45,300 LBS
SECOND STAGE 57 1/2 FT
W_P = 103,200 LBS CAPACITY
W_G = 157,600
T/W = 1.27
168 DIA (14 FT)
LH_2 TANK
4180 CU FT
LO_2 TANK
1270 CU FT
ROLL CONTROL
4 ROCKETS WITH
TANGENTIAL THRUST
J-2
W_G = 767400 LBS

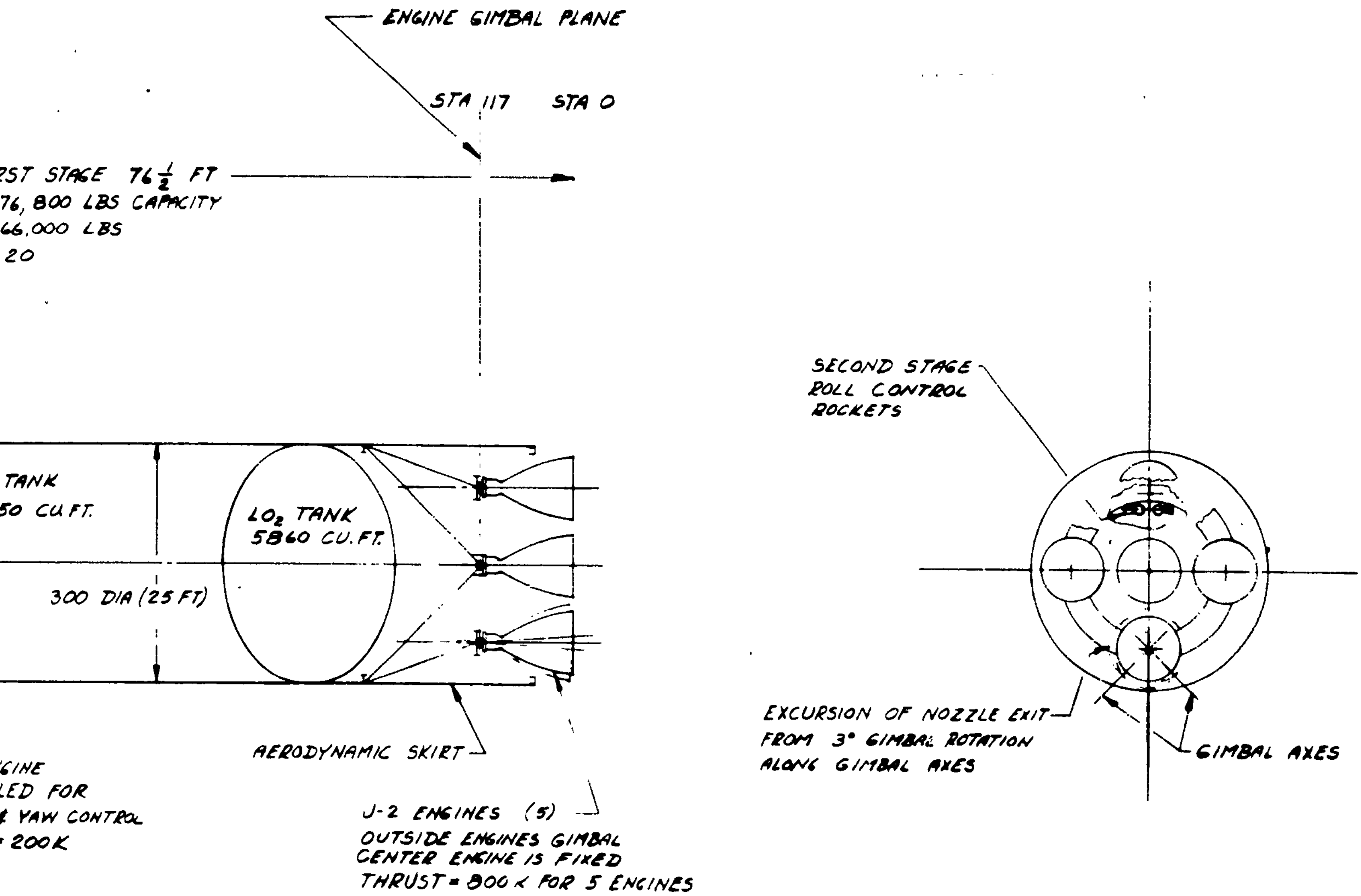
1

SID 61-341



2

Figure A-16. 800K J-2 First Stage - J-

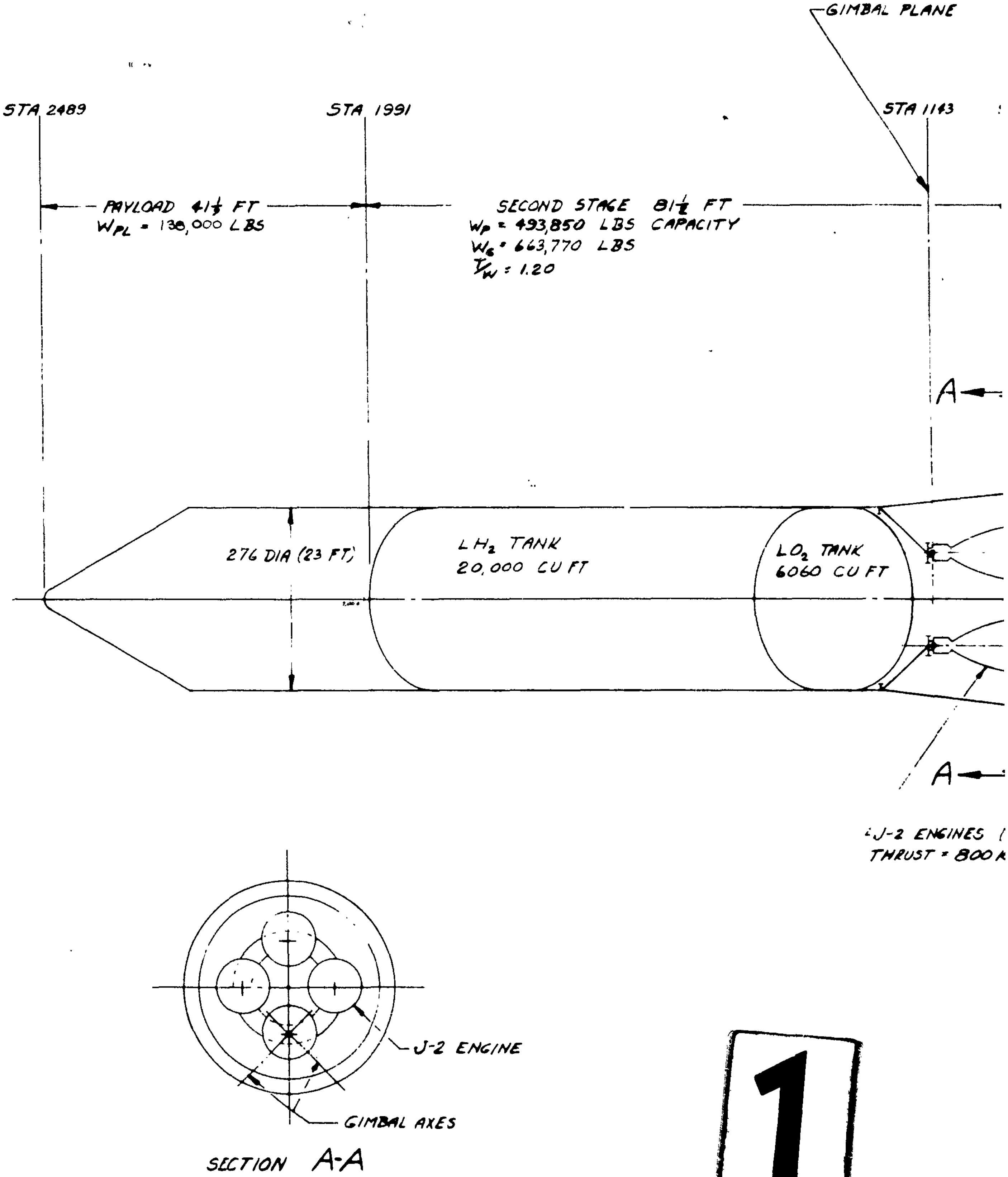


3

Figure A-16. 800K J-2 First Stage - J-2 Second Stage

[redacted]

[redacted]



1

SID 61-341

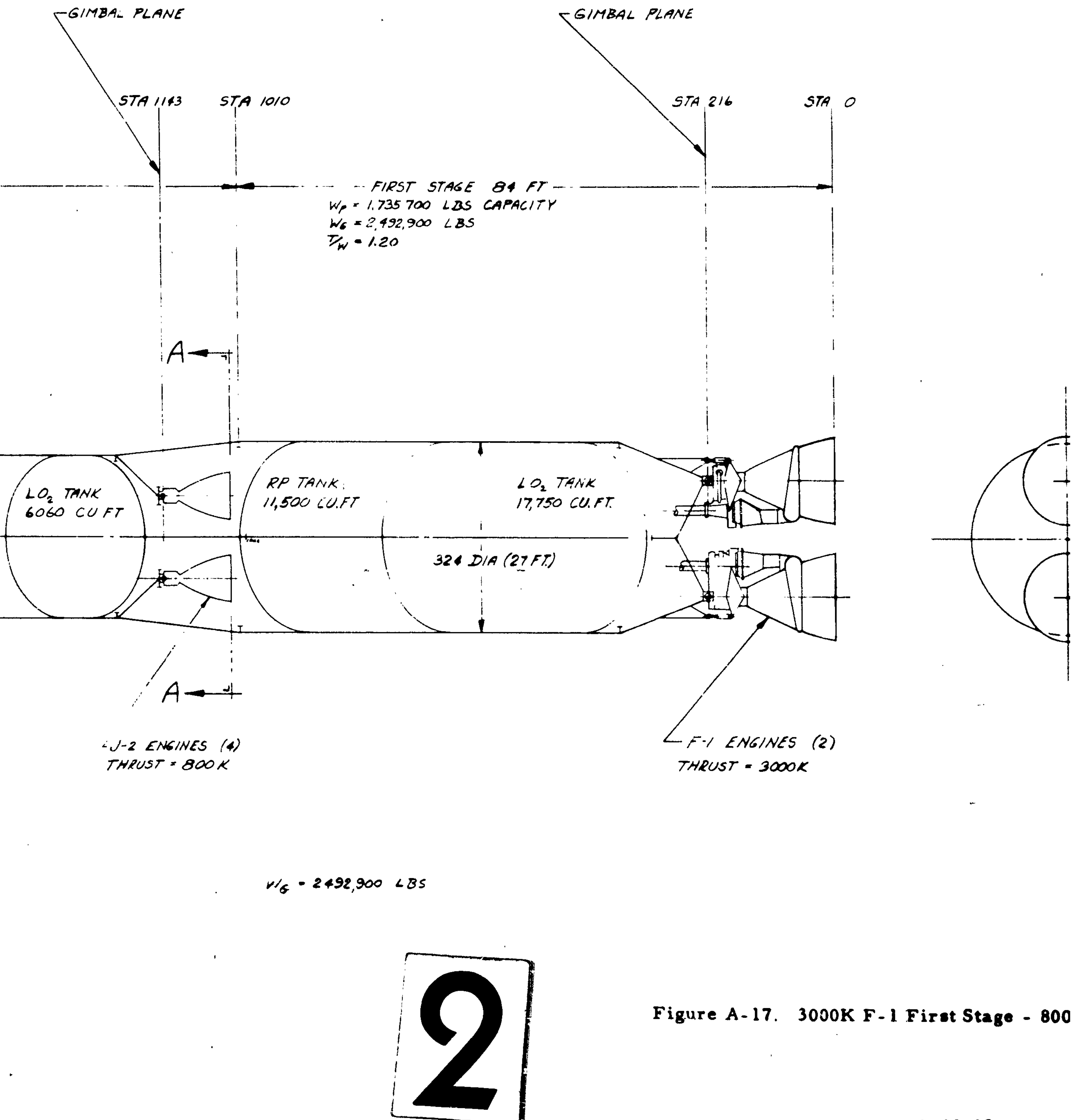
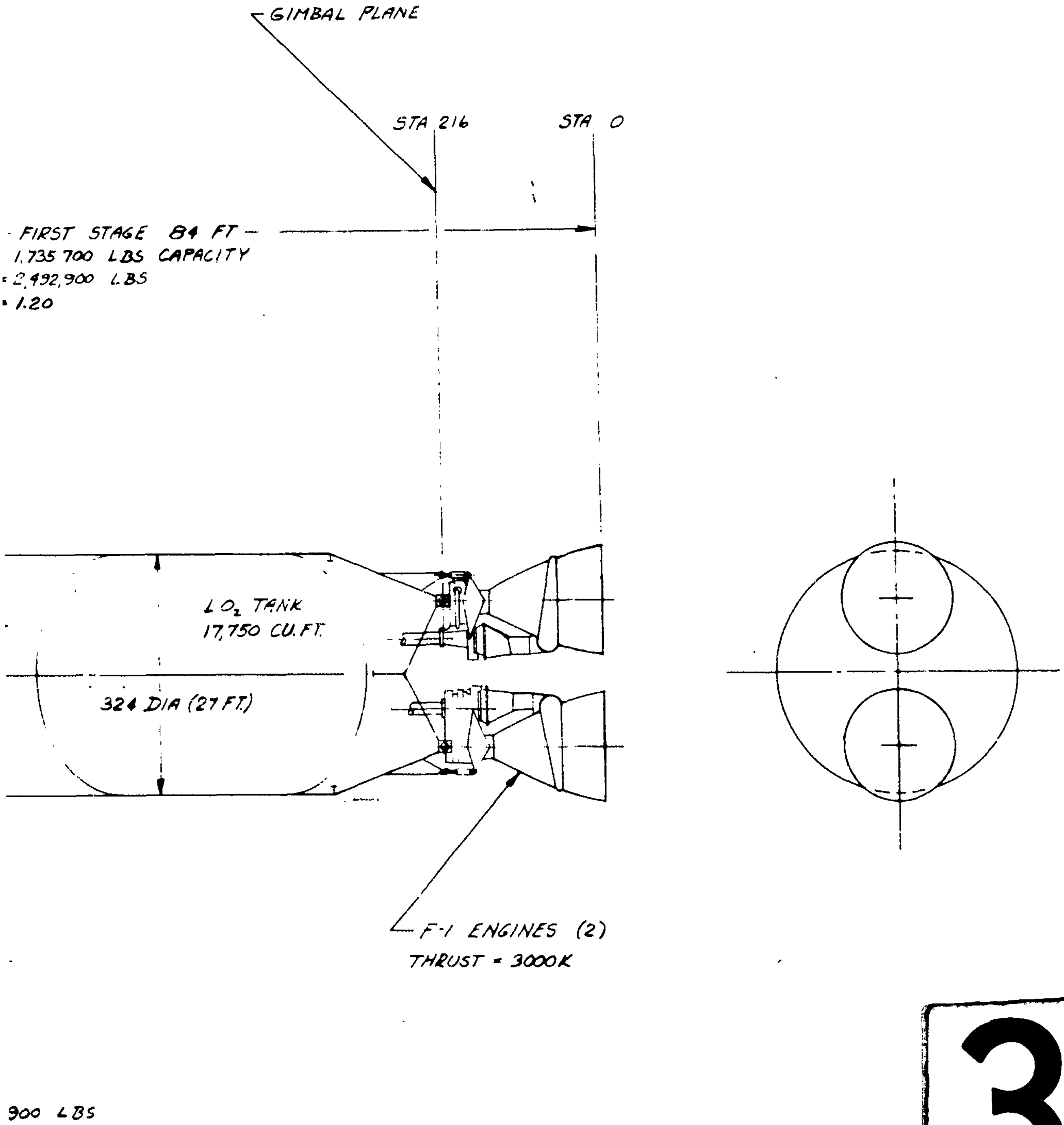


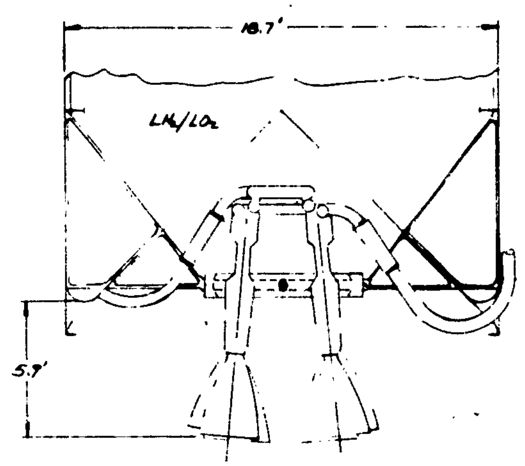
Figure A-17. 3000K F-1 First Stage - 800



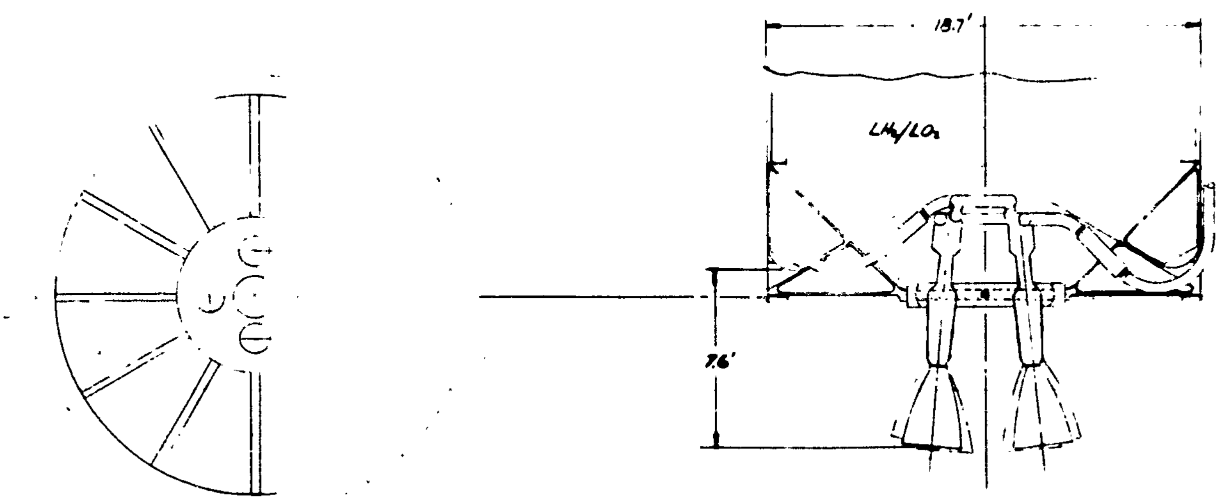
3

Figure A-17. 3000K F-1 First Stage - 800K J-2 Second Stage

[redacted]



4-150 K' ENGINES GIMBALED AS A UNIT
TANK BULKHEAD-45 DEG CONCIAL
INVERTED: ENGINE MT STRUCTURE-
RADIAL BEAMS

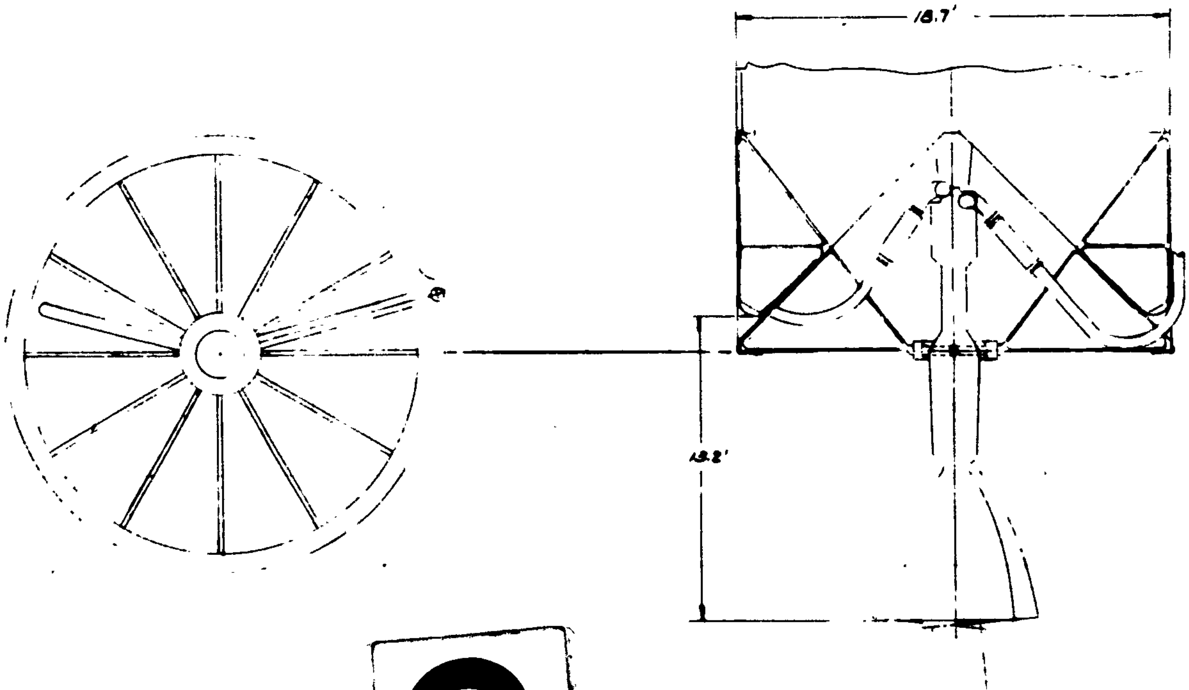


4-150 K' ENGINES GIMBALED AS A UNIT
TANK BULKHEAD-30 DEG. CONCIAL
INVERTED: ENGINE MT STRUCTURE-
RADIAL BEAMS.

1

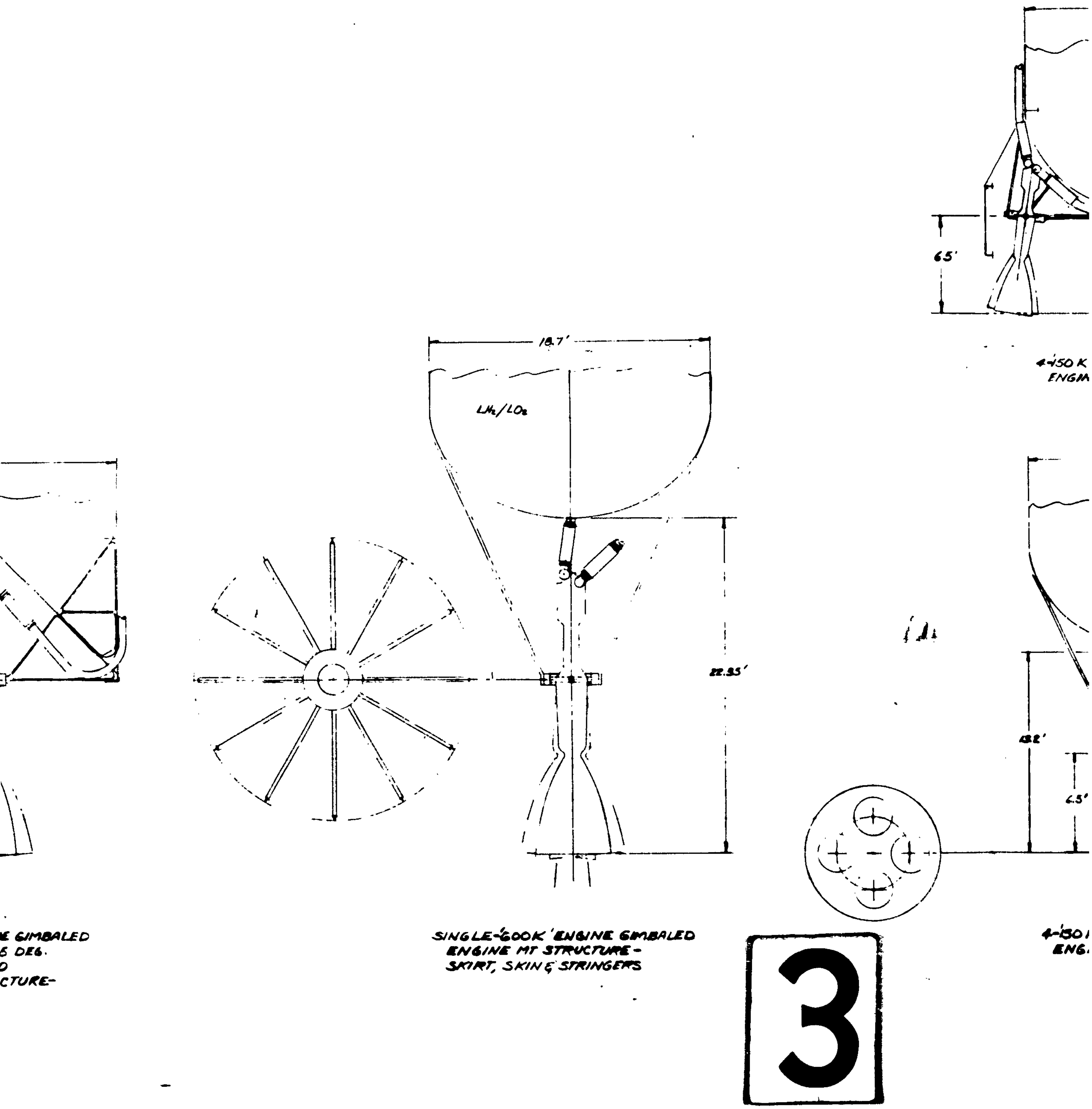
UNIT
RE-

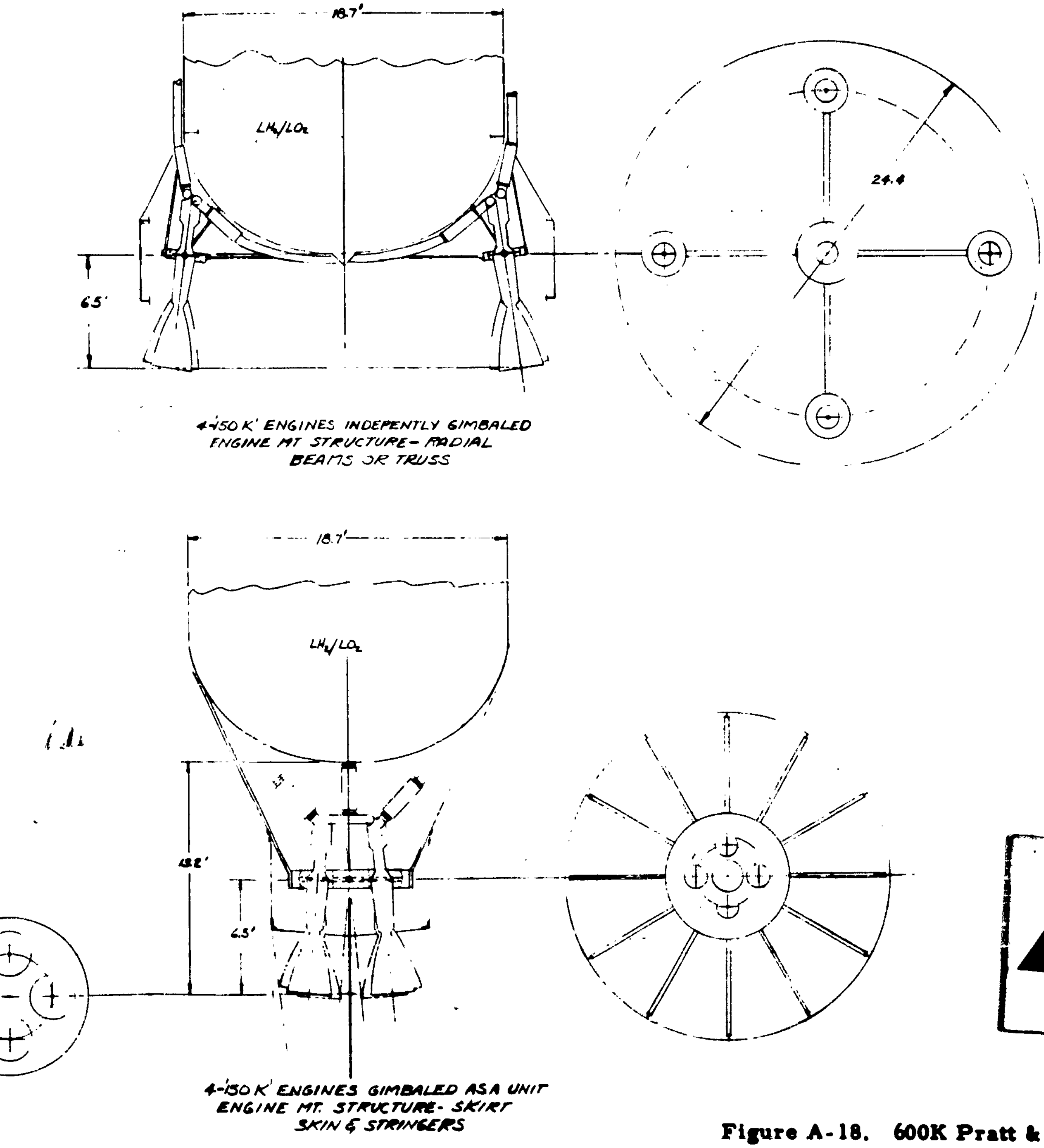
IAL
TURE-



2

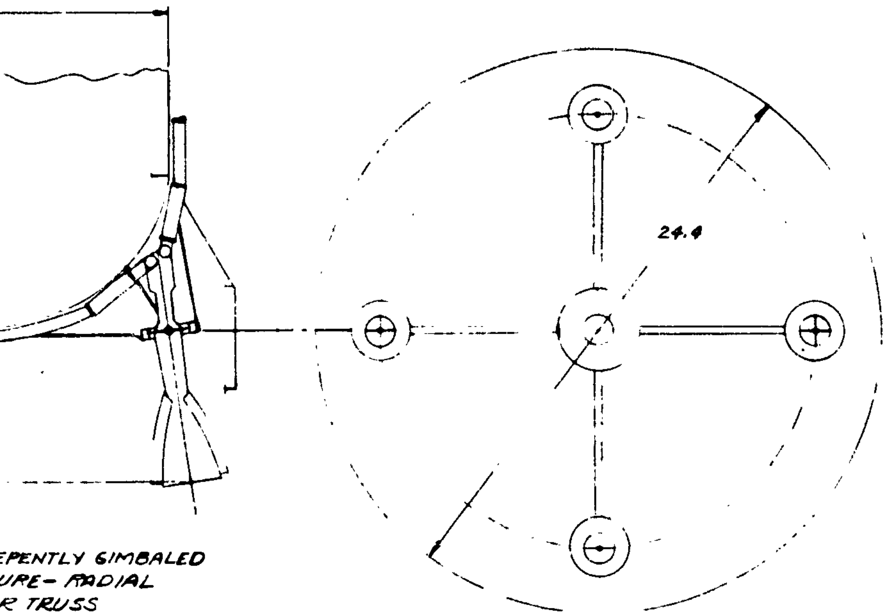
SINGLE-'600K' ENGINE GIMBALED
TANK BULKHEAD-45 DEG.
CONCIAL INVERTED
ENGINE MT. STRUCTURE-
RADIAL BEAMS

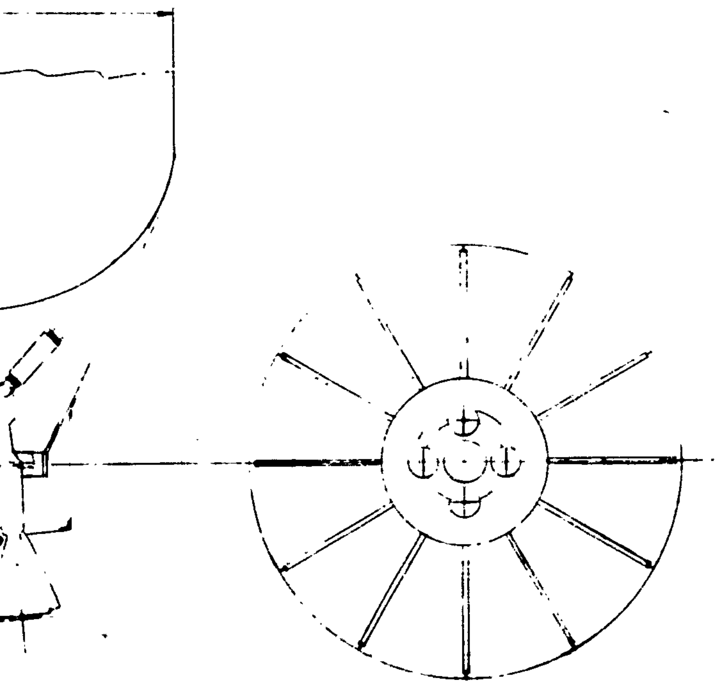

18.7'
LH2/LO2
22.35'
65'
4-150 K
ENGIN
13.2'
6.5'
E GIMBALED
5 DEG.
D
CTURE-
SINGLE-600K ENGINE GIMBALED
ENGINE MT STRUCTURE-
SKIRT, SKIN & STRINGERS
4-150
ENG.
3



4

Figure A-18. 600K Pratt & Whitney Engine Ins





IMBALED ASA UNIT
CTURE- SKIRT
TRINGERS

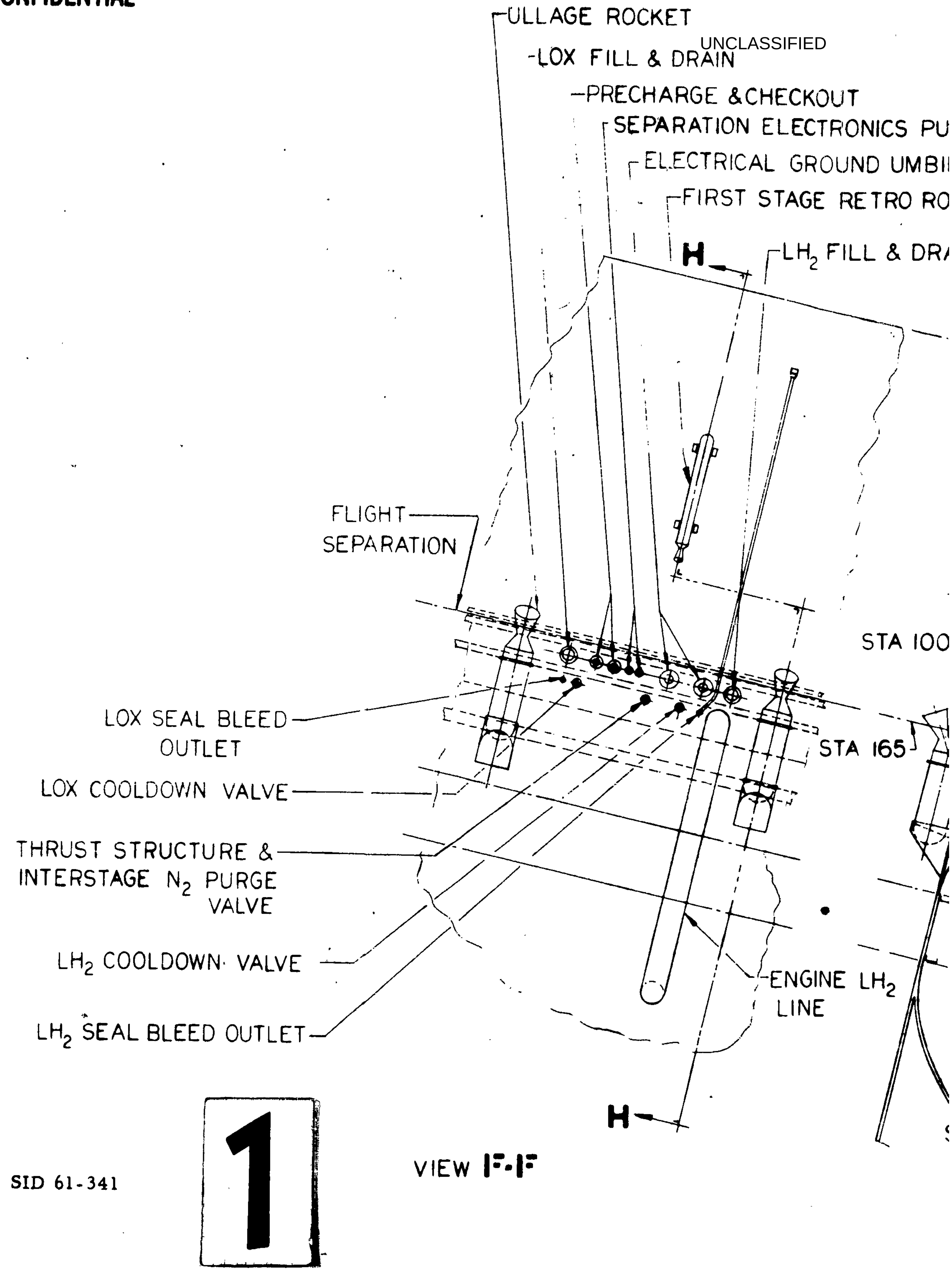
5

Figure A-18. 600K Pratt & Whitney Engine Installation Study

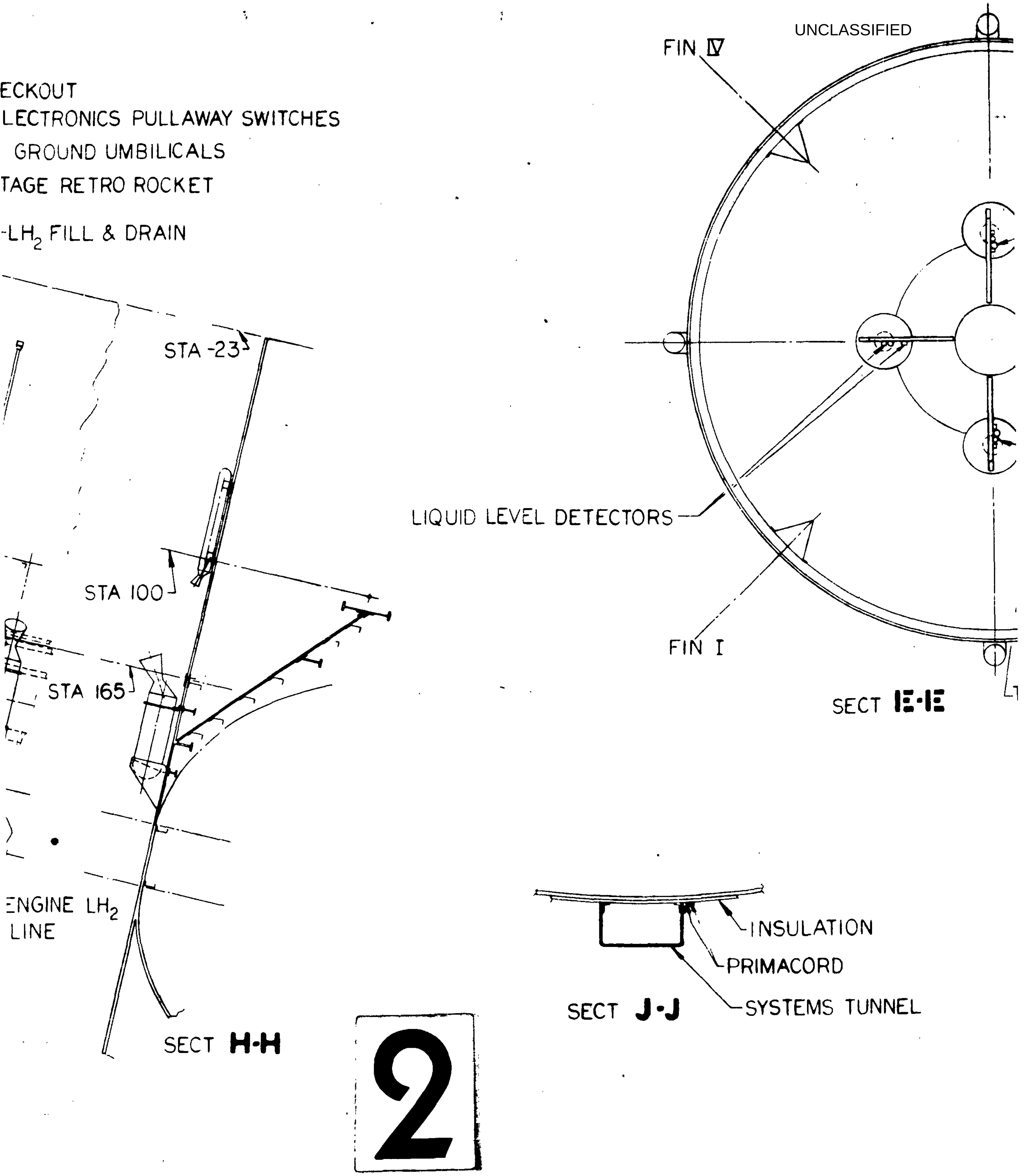
[redacted]

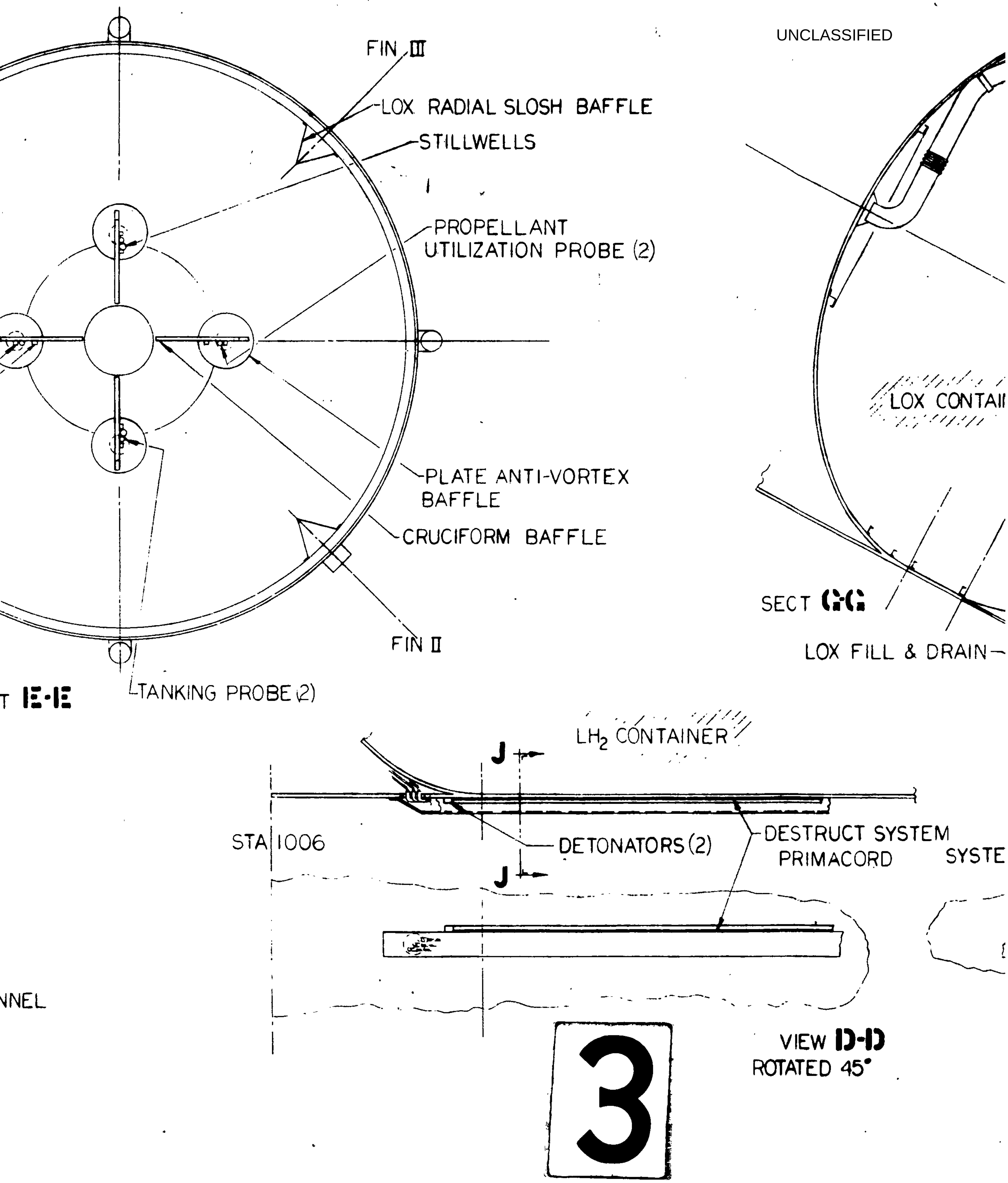
CONFIDENTIAL

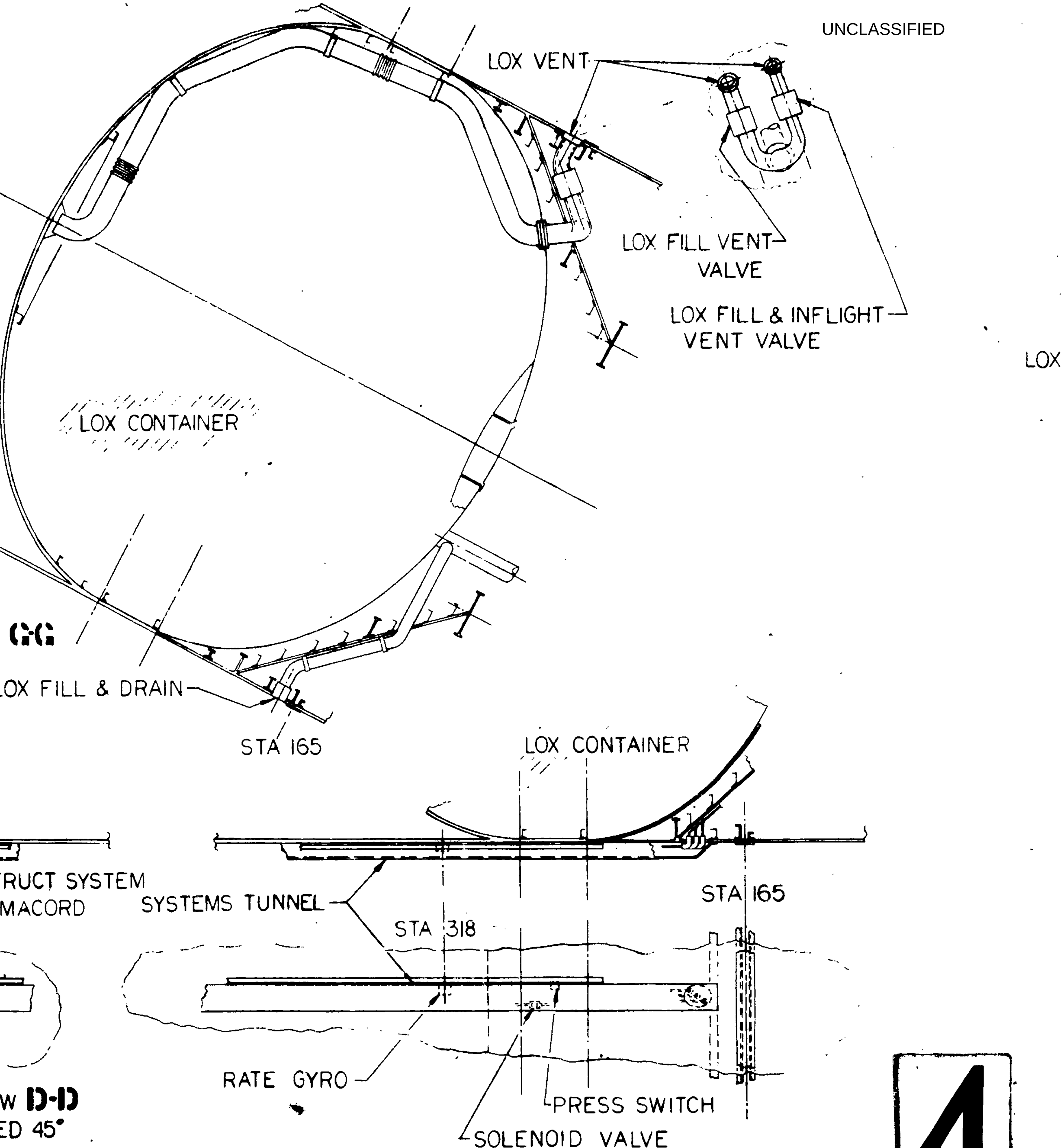
UNCLASSIFIED



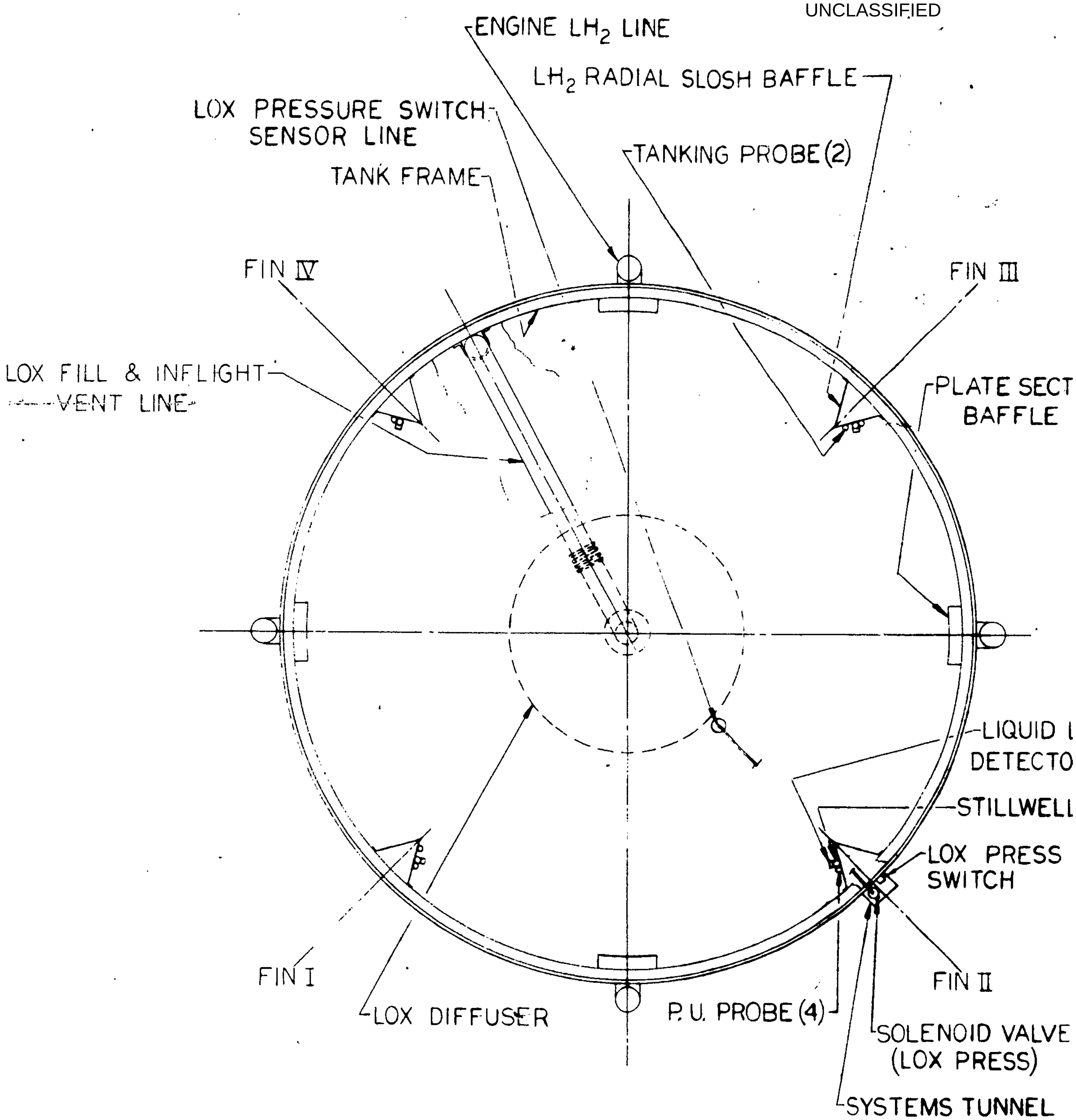
SID 61-341


ECKOUT
LECTRONICS PULLAWAY SWITCHES
GROUND UMBILICALS
TAGE RETRO ROCKET
-LH_2 FILL & DRAIN
STA -23
STA 100
STA 165
ENGINE LH_2 LINE
SECT H-H
FIN IV
LIQUID LEVEL DETECTORS
FIN I
SECT E-E
INSULATION
PRIMACORD
SYSTEMS TUNNEL
SECT J-J
2


FIN III
LOX RADIAL SLOSH BAFFLE
STILLWELLS
PROPELLANT UTILIZATION PROBE (2)
PLATE ANTI-VORTEX BAFFLE
CRUCIFORM BAFFLE
FIN II
TANKING PROBE (2)
E-E
LOX CONTAI
SECT G-G
LOX FILL & DRAIN
LH_2 CONTAINER
J
STA 1006
DETONATORS (2)
DESTRUCT SYSTEM PRIMACORD
SYSTE
J
NNEL
VIEW D-D
ROTATED 45°
3

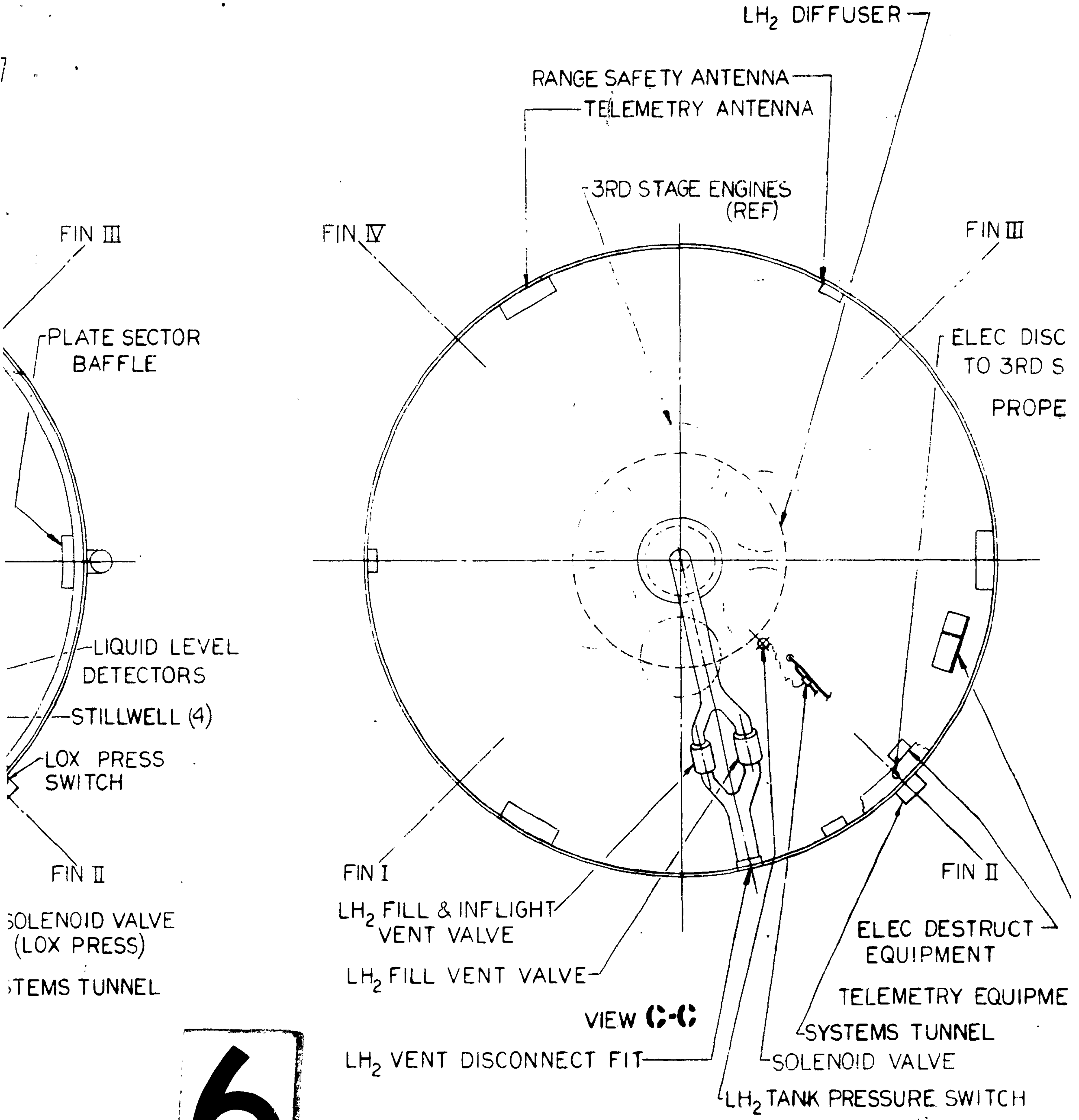

LOX VENT
LOX FILL VENT
VALVE
LOX FILL & INFLIGHT
VENT VALVE
LOX
LOX CONTAINER
G-G
LOX FILL & DRAIN
STA 165
LOX CONTAINER
TRUCT SYSTEM
MACORD
SYSTEMS TUNNEL
STA 165
STA 318
RATE GYRO
W D-D
ED 45°
PRESS SWITCH
SOLENOID VALVE

4


ENGINE LH_2 LINE
LH_2 RADIAL SLOSH BAFFLE
LOX PRESSURE SWITCH SENSOR LINE
TANKING PROBE (2)
TANK FRAME
FIN IV
FIN III
LOX FILL & INFLIGHT VENT LINE
PLATE SECT BAFFLE
LIQUID L DETECTO
STILLWELL
LOX PRESS SWITCH
FIN I
FIN II
LOX DIFFUSER
P. U. PROBE (4)
SOLENOID VALVE (LOX PRESS)
SYSTEMS TUNNEL

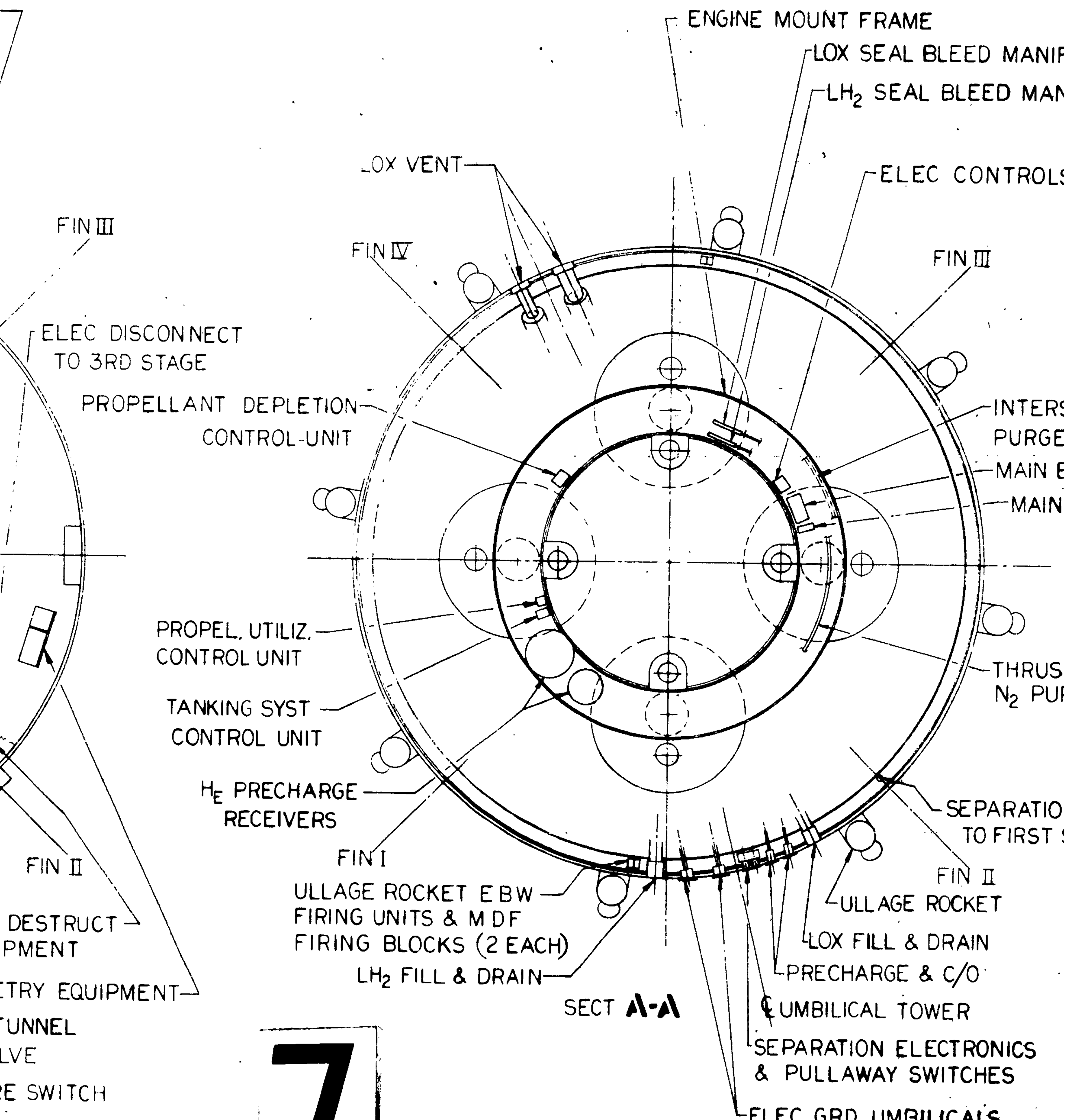
SECT 13-13

5

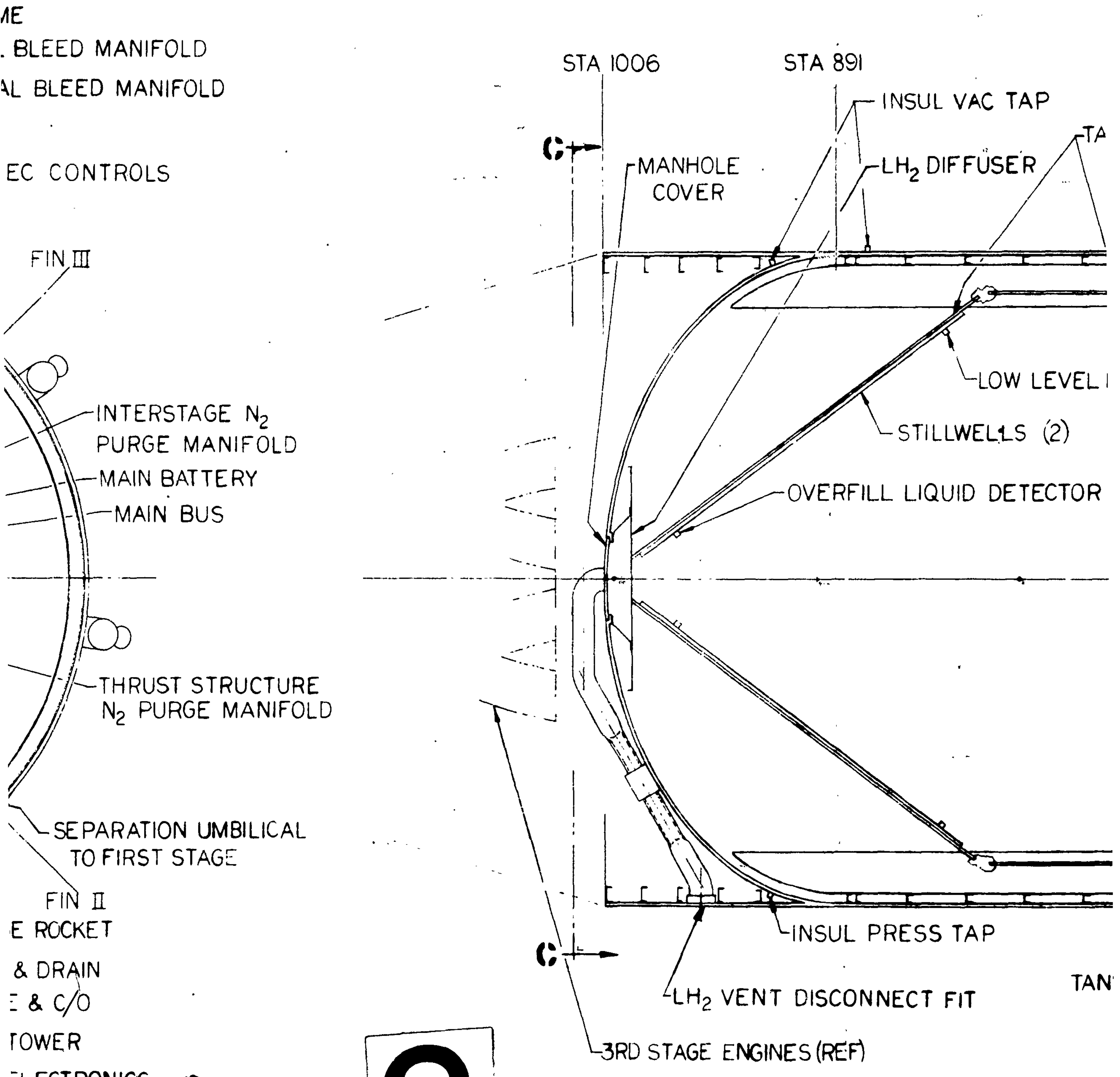

LH2 DIFFUSER
RANGE SAFETY ANTENNA
TELEMETRY ANTENNA
3RD STAGE ENGINES (REF)
FIN III
FIN IV
FIN III
PLATE SECTOR BAFFLE
ELEC DISC
TO 3RD S
PROPE
LIQUID LEVEL DETECTORS
STILLWELL (4)
LOX PRESS SWITCH
FIN II
FIN I
FIN II
SOLENOID VALVE (LOX PRESS)
STEMS TUNNEL
LH2 FILL & INFLIGHT VENT VALVE
LH2 FILL VENT VALVE
ELEC DESTRUCT EQUIPMENT
TELEMETRY EQUIPME
SYSTEMS TUNNEL
SOLENOID VALVE
LH2 VENT DISCONNECT FIT
LH2 TANK PRESSURE SWITCH

VIEW C-C

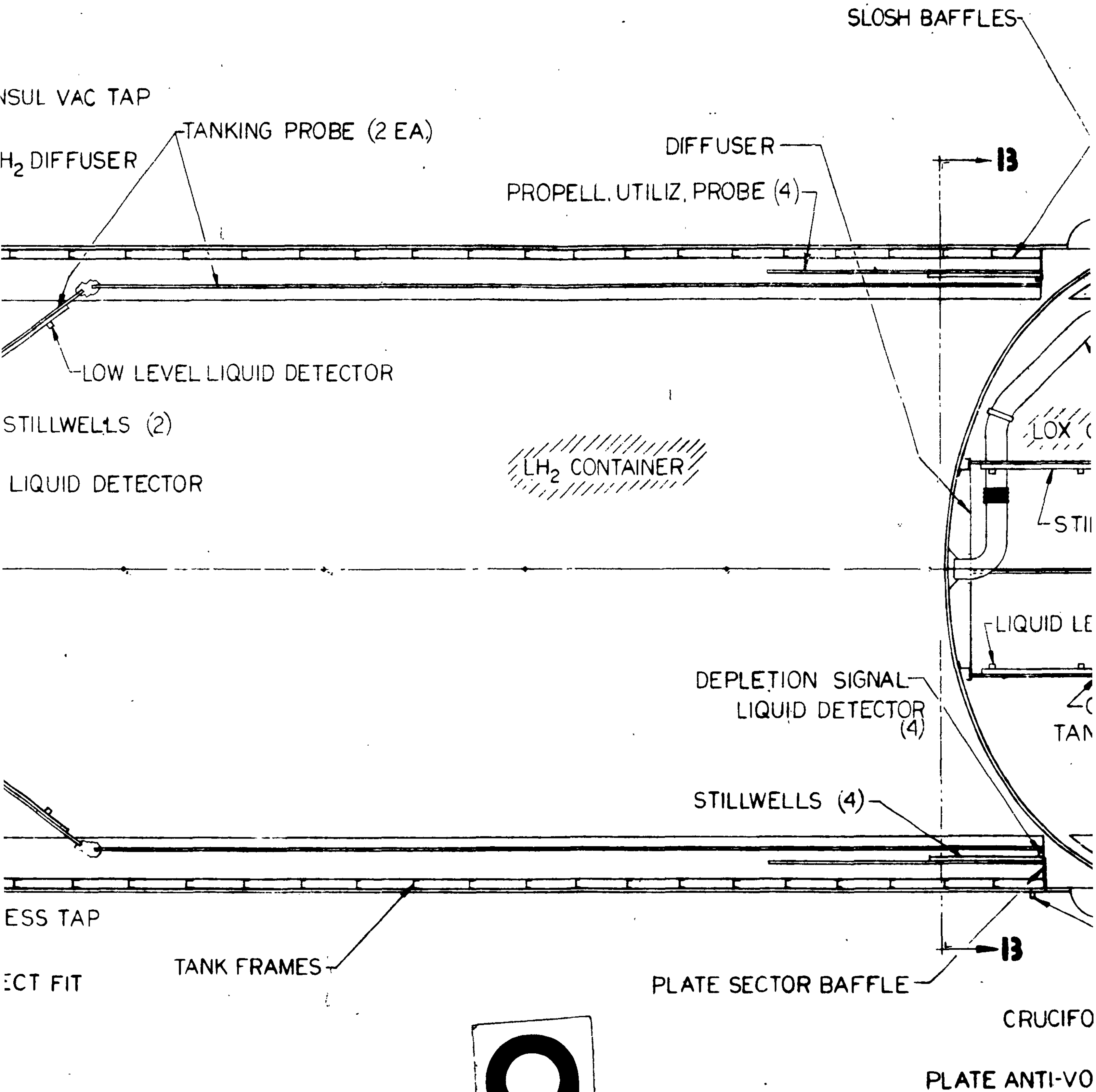
6


ENGINE MOUNT FRAME
LOX SEAL BLEED MANIF
LH2 SEAL BLEED MAN
LOX VENT
ELEC CONTROLS
FIN III
FIN IV
FIN III
ELEC DISCONNECT
TO 3RD STAGE
PROPELLANT DEPLETION
CONTROL-UNIT
INTERS
PURGE
MAIN E
MAIN
PROPEL. UTILIZ.
CONTROL UNIT
THRUS
N2 PU
TANKING SYST
CONTROL UNIT
HE PRECHARGE
RECEIVERS
SEPARATIO
TO FIRST
FIN I
FIN II
FIN II
ULLAGE ROCKET EBW
FIRING UNITS & MDF
FIRING BLOCKS (2 EACH)
ULLAGE ROCKET
DESTRUCT
PMENT
LOX FILL & DRAIN
LH2 FILL & DRAIN
PRECHARGE & C/O
TRY EQUIPMENT
SECT A-A
℄ UMBILICAL TOWER
TUNNEL
LVE
SEPARATION ELECTRONICS
& PULLAWAY SWITCHES
E SWITCH
ELEC GRD UMBILICALS

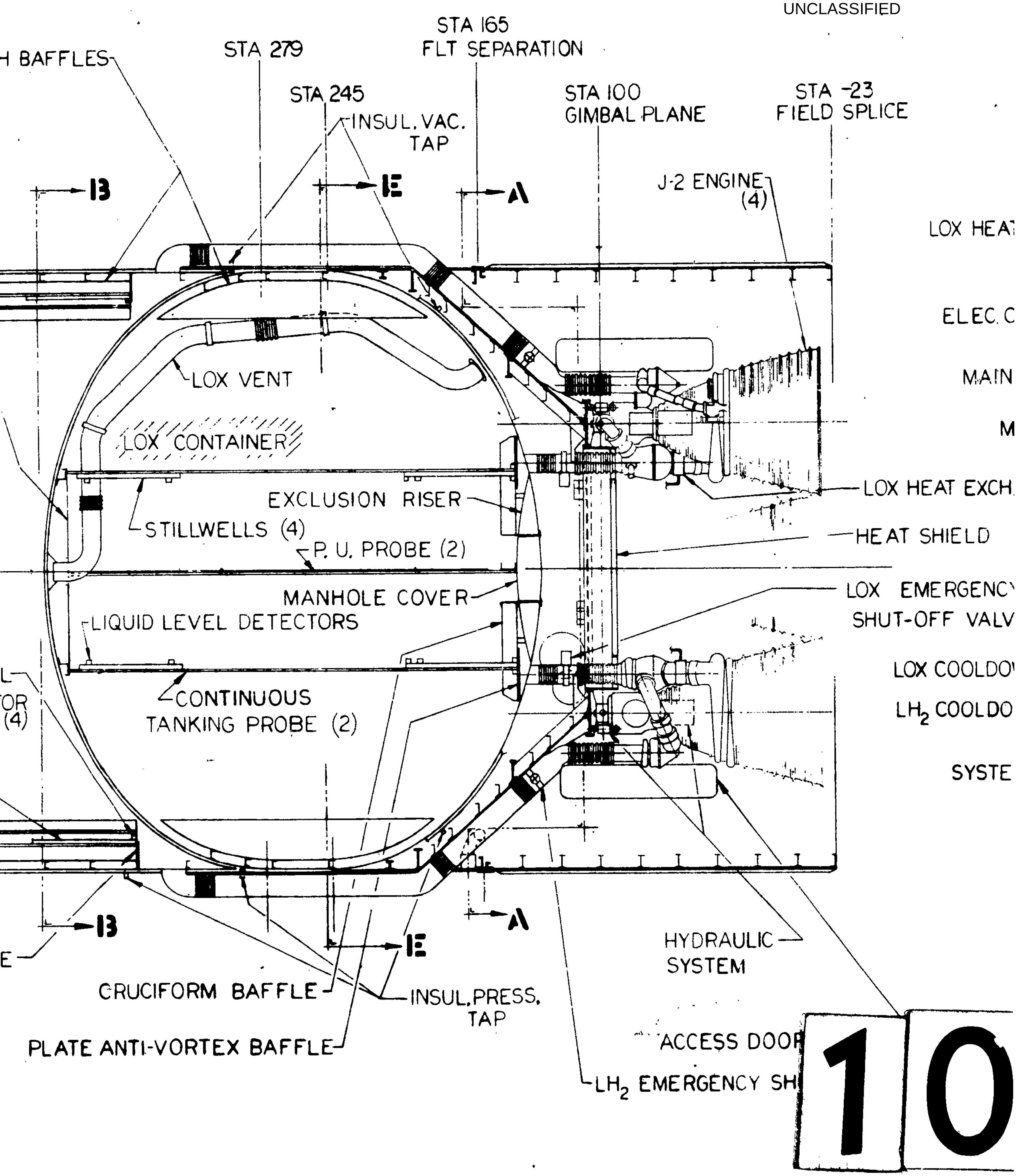
7


ME
. BLEED MANIFOLD
AL BLEED MANIFOLD
EC CONTROLS
FIN III
INTERSTAGE N_2 PURGE MANIFOLD
MAIN BATTERY
MAIN BUS
THRUST STRUCTURE N_2 PURGE MANIFOLD
SEPARATION UMBILICAL TO FIRST STAGE
FIN II
E ROCKET
& DRAIN
E & C/O
TOWER
ELECTRONICS
SWITCHES
BILICALS
STA 1006
STA 891
INSUL VAC TAP
MANHOLE COVER
LH_2 DIFFUSER
LOW LEVEL
STILLWELLS (2)
OVERFILL LIQUID DETECTOR
INSUL PRESS TAP
LH_2 VENT DISCONNECT FIT
3RD STAGE ENGINES (REF)
C
C

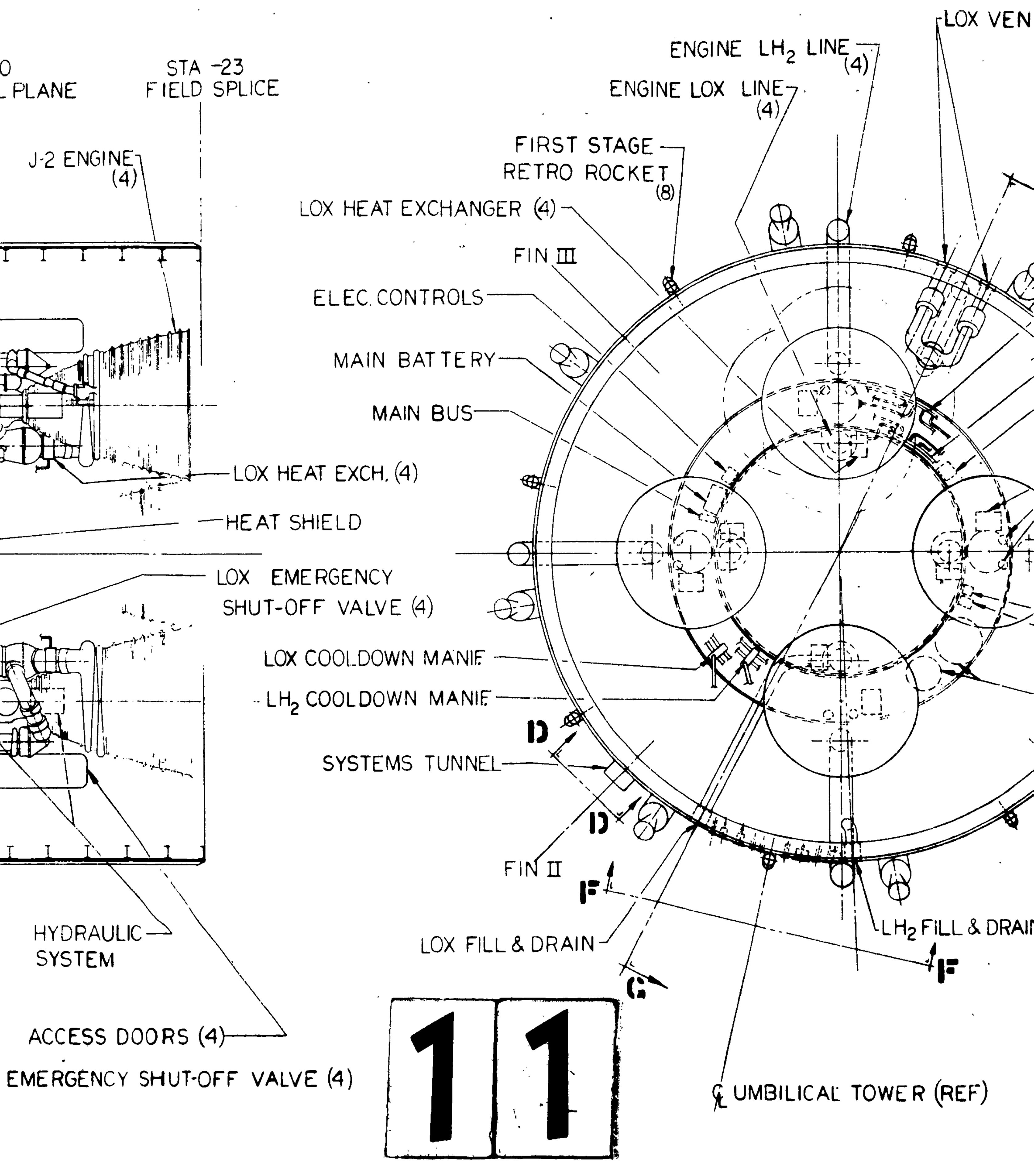
8


SLOSH BAFFLES
NSUL VAC TAP
TANKING PROBE (2 EA)
H_2 DIFFUSER
DIFFUSER
PROPELL. UTILIZ. PROBE (4)
B
LOW LEVEL LIQUID DETECTOR
STILLWELLS (2)
LIQUID DETECTOR
LH_2 CONTAINER
LOX
STI
LIQUID LE
DEPLETION SIGNAL
LIQUID DETECTOR
(4)
TAN
STILLWELLS (4)
ESS TAP
TANK FRAMES
B
ECT FIT
PLATE SECTOR BAFFLE
CRUCIFO
PLATE ANTI-VO

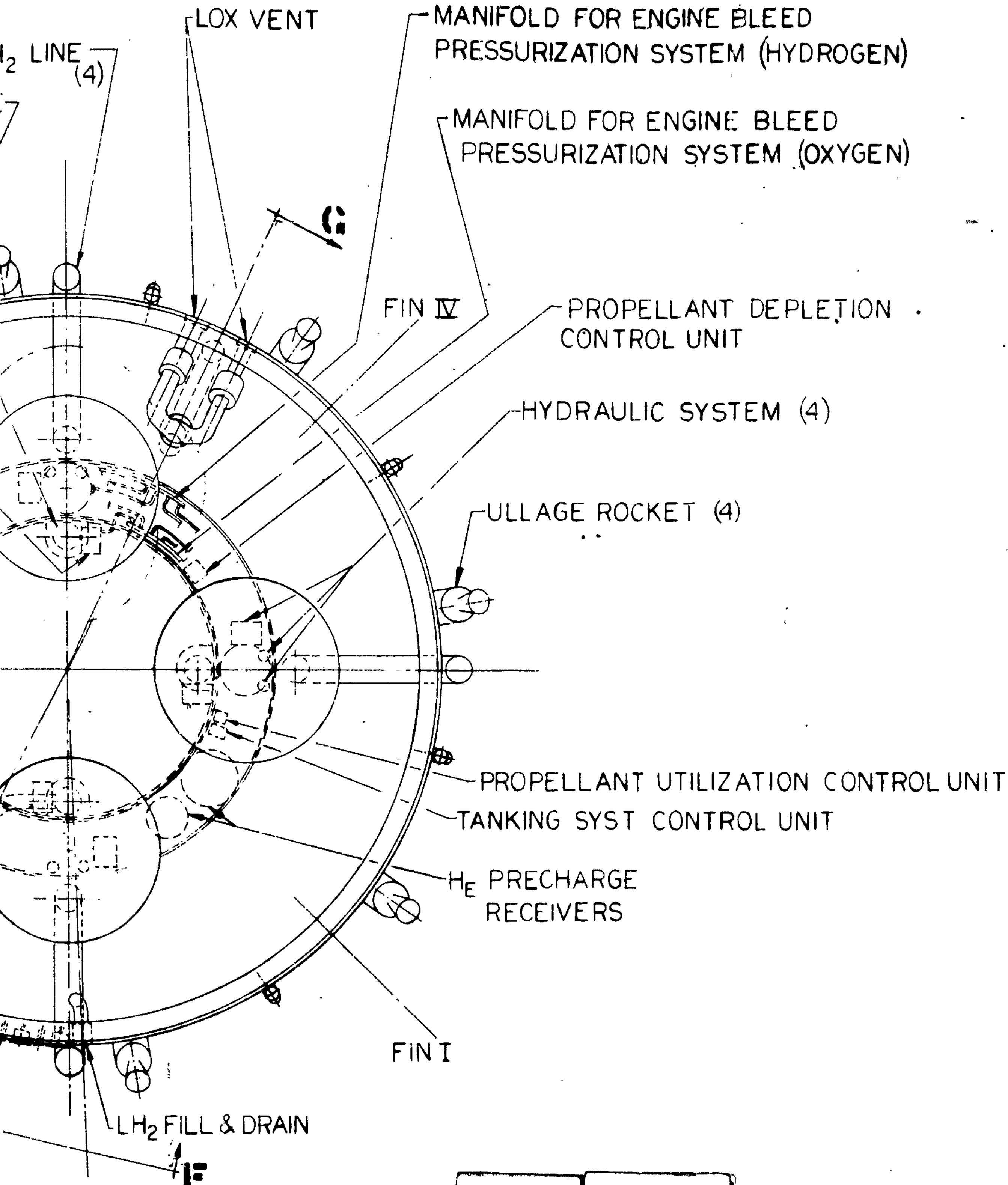
9



H BAFFLES
STA 279
STA 165
FLT SEPARATION
STA 245
INSUL. VAC. TAP
STA 100
GIMBAL PLANE
STA -23
FIELD SPLICE
B
E
A
J-2 ENGINE (4)
LOX HEA
ELEC. C
MAIN
M
LOX VENT
LOX CONTAINER
EXCLUSION RISER
STILLWELLS (4)
P. U. PROBE (2)
LOX HEAT EXCH.
HEAT SHIELD
MANHOLE COVER
LIQUID LEVEL DETECTORS
LOX EMERGENCY
SHUT-OFF VALV
LOX COOLDO
LH_2 COOLDO
SYSTE
CONTINUOUS
TANKING PROBE (2)
L
OR
(4)
HYDRAULIC
SYSTEM
E
CRUCIFORM BAFFLE
INSUL. PRESS. TAP
PLATE ANTI-VORTEX BAFFLE
ACCESS DOOR
LH_2 EMERGENCY SH
10



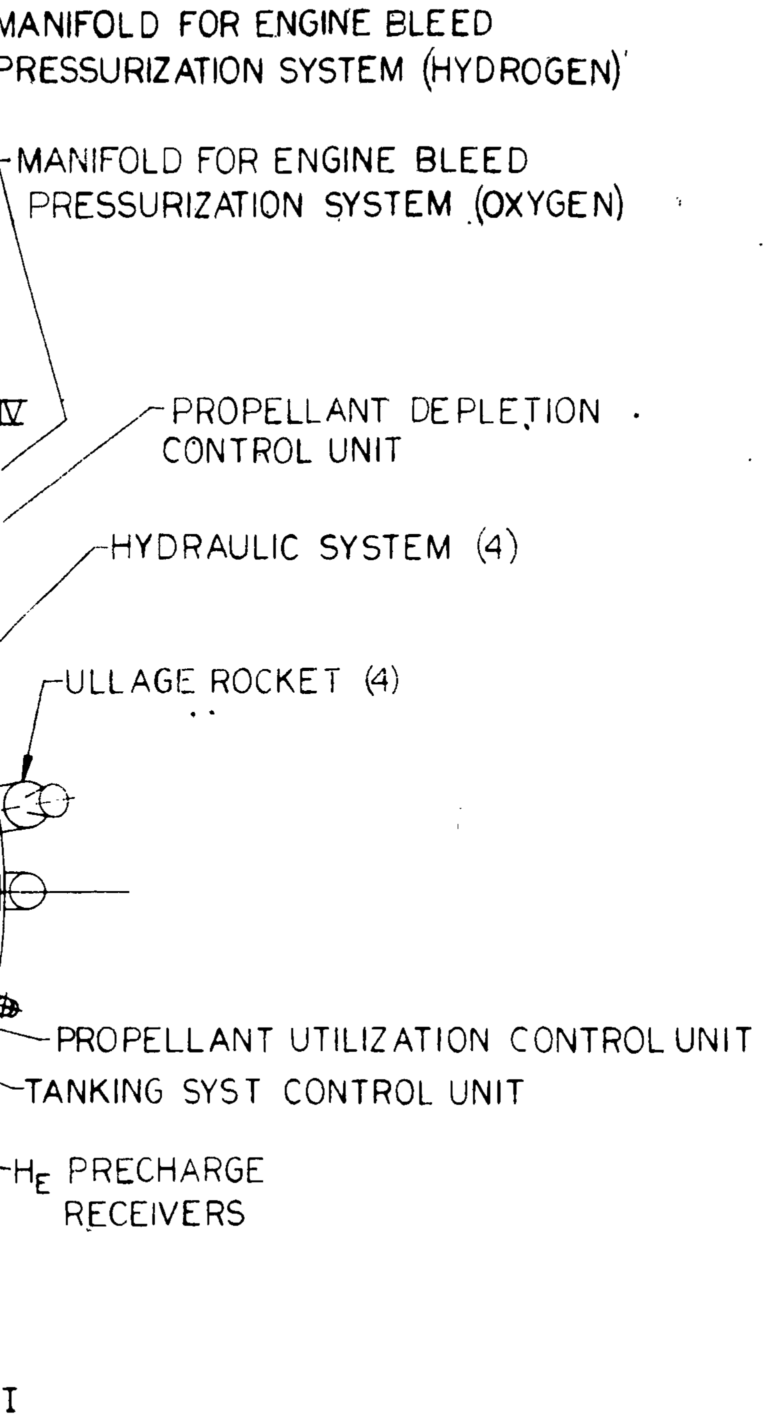
LOX VEN
ENGINE LH2 LINE (4)
ENGINE LOX LINE (4)
0
PLANE
STA -23
FIELD SPLICE
J-2 ENGINE (4)
FIRST STAGE RETRO ROCKET (8)
LOX HEAT EXCHANGER (4)
FIN III
ELEC. CONTROLS
MAIN BATTERY
MAIN BUS
LOX HEAT EXCH. (4)
HEAT SHIELD
LOX EMERGENCY SHUT-OFF VALVE (4)
LOX COOLDOWN MANIF.
LH2 COOLDOWN MANIF.
D
SYSTEMS TUNNEL
D
FIN II
F
LH2 FILL & DRAI
LOX FILL & DRAIN
F
G
HYDRAULIC SYSTEM
ACCESS DOORS (4)
EMERGENCY SHUT-OFF VALVE (4)
11
℄ UMBILICAL TOWER (REF)



12

Figure A-19. J-2 Installati

A-85, 86

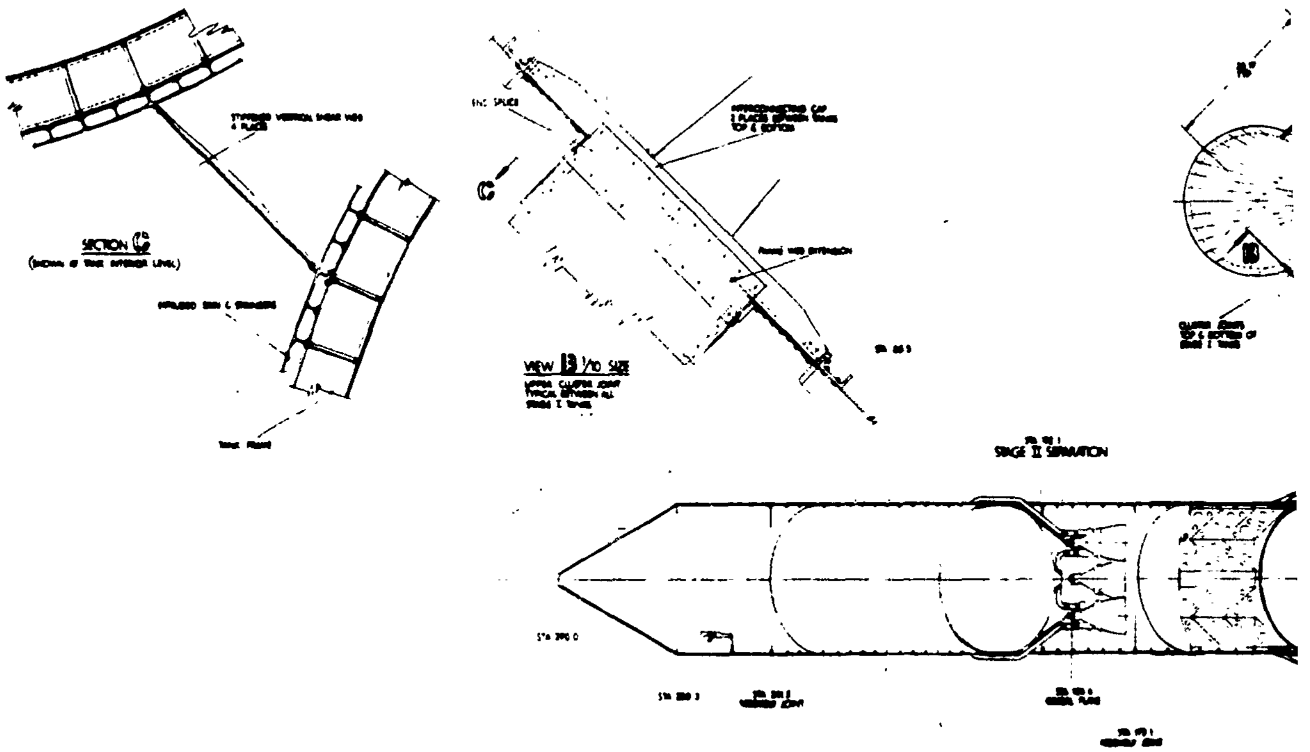


13

Figure A-19. J-2 Installation (S-II)

[redacted]

[redacted]



PAYLOAD 137,500

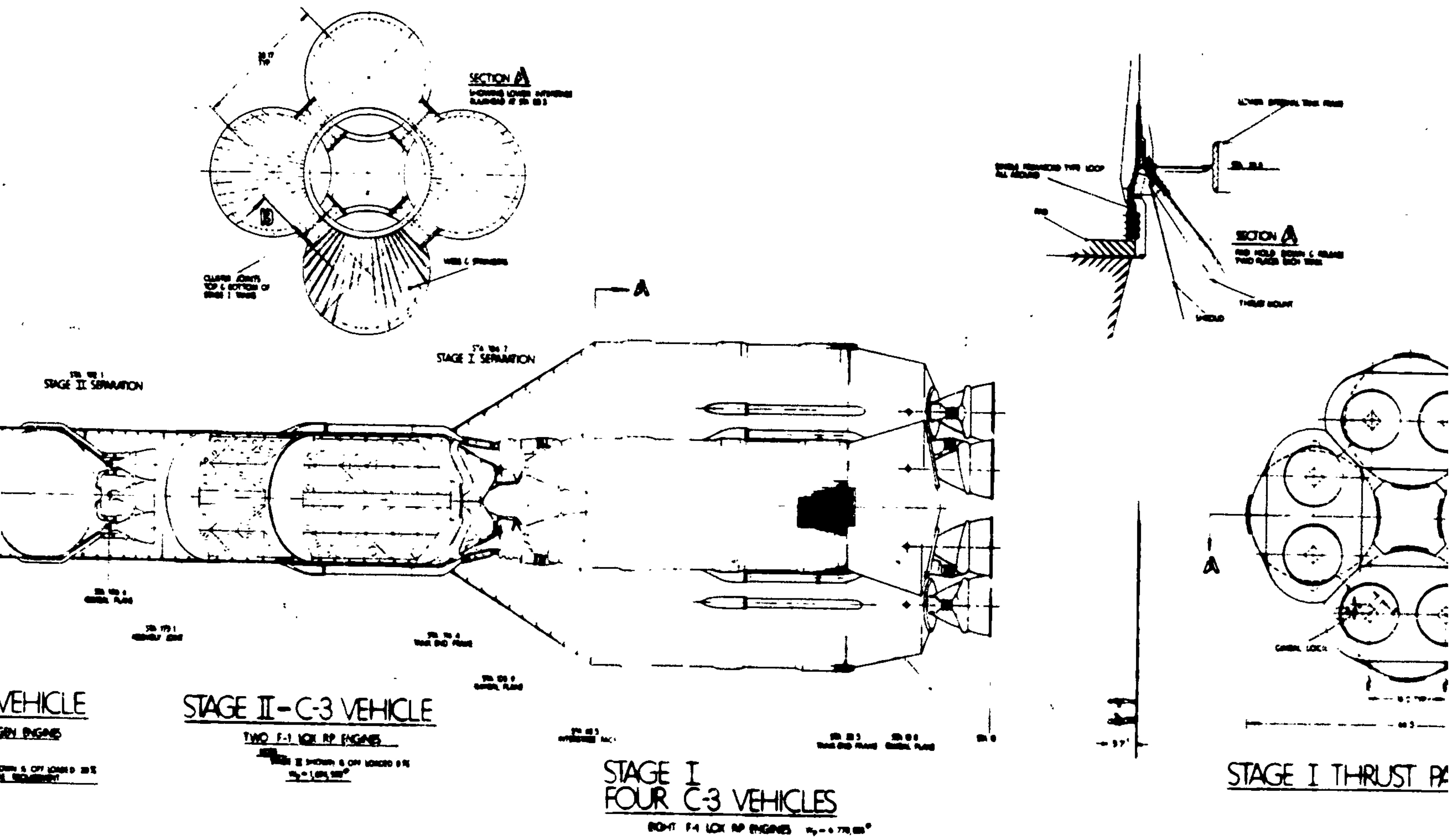
STAGE III - S-II VEHICLE

FOUR J-2 LOX HYDROGEN ENGINES

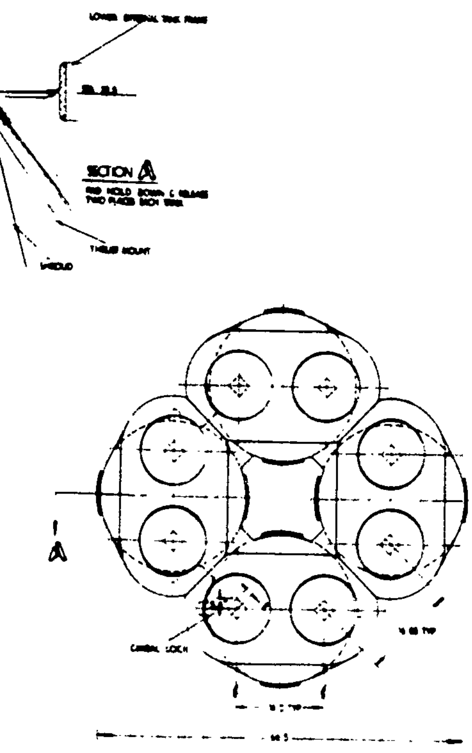
STAGE II -

1

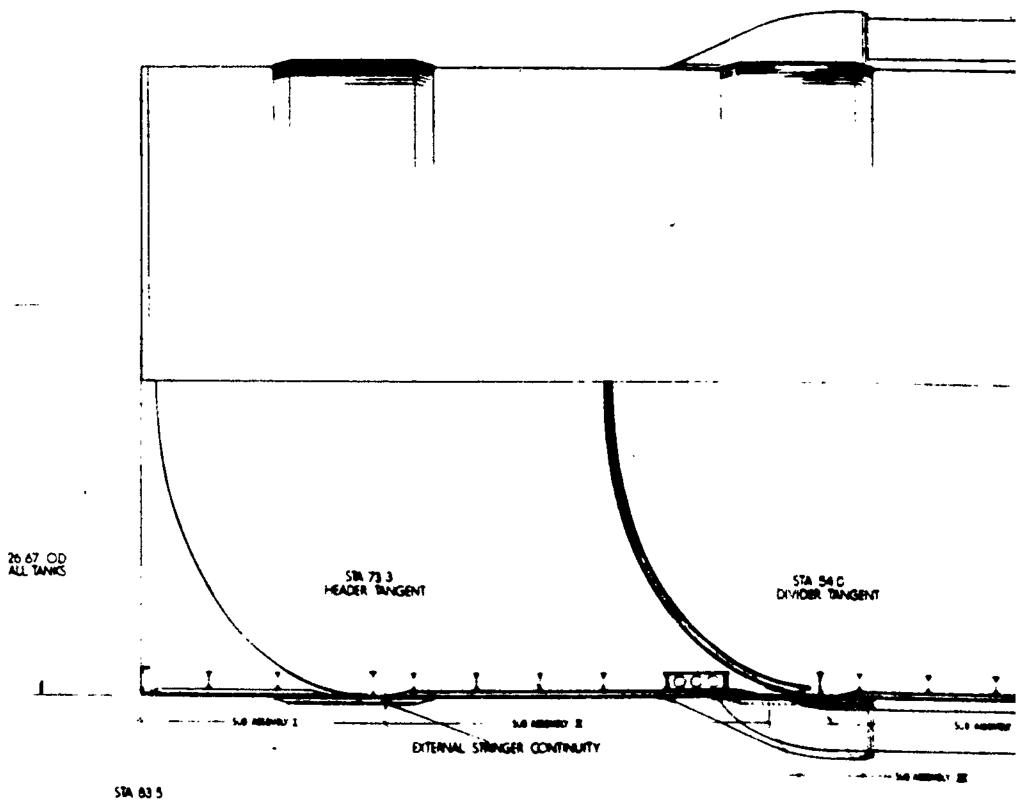
SID 61-341


SECTION A
SECTION A
A
STAGE I SEPARATION
STAGE II SEPARATION
VEHICLE
STAGE II - C-3 VEHICLE
TWO F-1 LOX RP ENGINES
STAGE I
FOUR C-3 VEHICLES
STAGE I THRUST

2



STAGE I THRUST PATTERN



C-3 MODULAR ELEMENT

(SLOSH BAFFLES OMITTED)

3

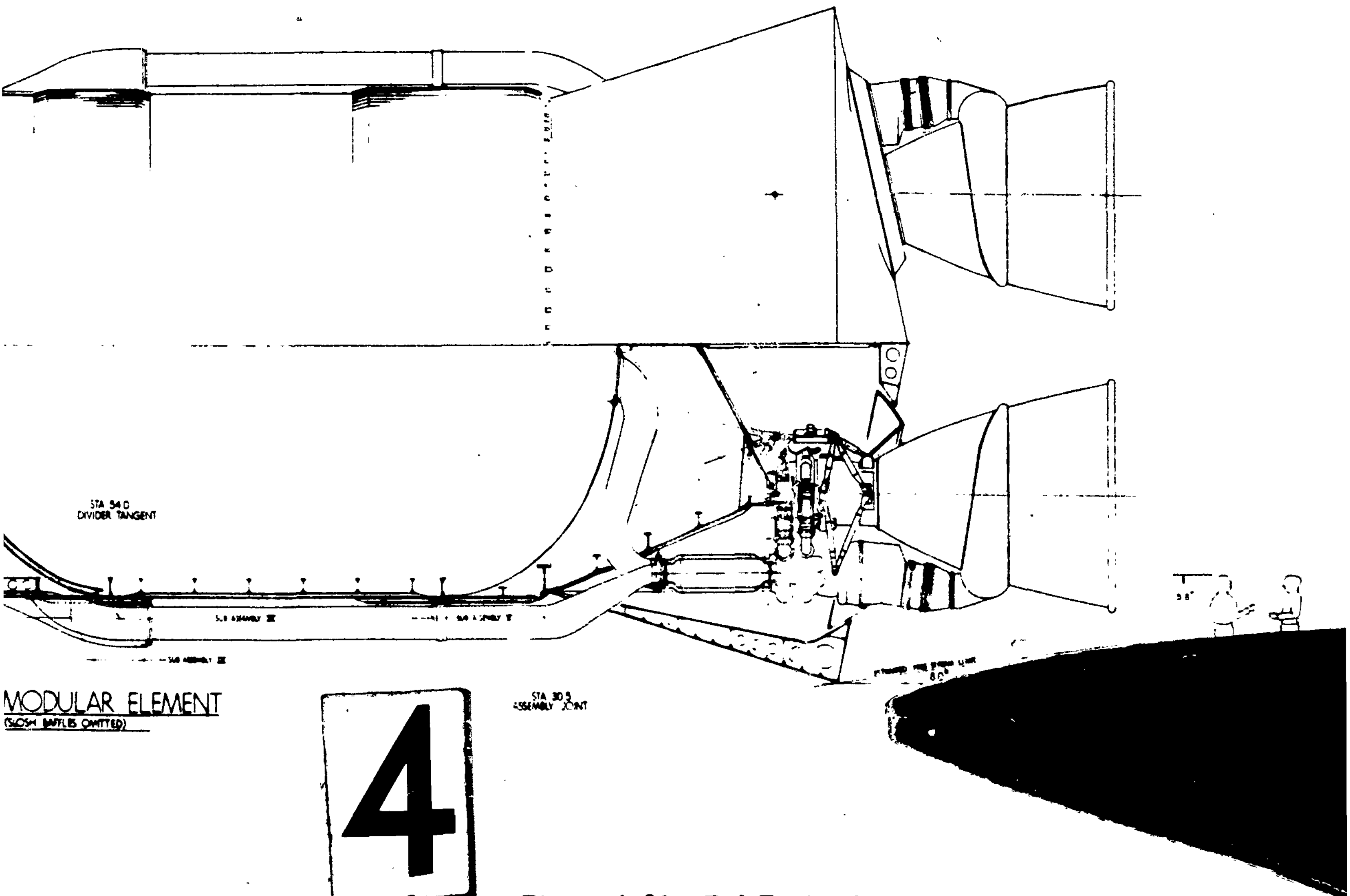


Figure A-20. F-1 Engine Installation (C-3)

[redacted]